NASA AGENA D
MISSION CAPABILITIES
AND RESTRAINTS CATALOG

Volume 2

N66-18426

FOREWORD

The NASA Agena D Mission Capabilities and Restraints Catalog has been assembled by the LeRC-Agena Project to delineate the procedures followed and the Agena D equipment used in integrating a payload with an Agena D-booster combination. The catalog consists of two volumes. Volume I is classified (CONF.) and contains information on the mission capabilities of the Agena D. Volume II contains equipment, programming, and procedural information.

This Volume II document is made up of three segments. The first segment contains material generated by the Lockheed Missiles & Space Company under contract NAS3-3805, Task 13 and the -2 revisions, as marked by the revision bars in the margins, are in accordance with Task 31 of the same contract. The second segment is the Agena Missions Standard Requirements and Restraints Document, I-Format and II-Instructions. The third segment is the LeRC Agena Project-Spacecraft Center Interface Operating Procedures.

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INTRODUCTION: VOLUME II

Volume I of the NASA Agena D Mission Capabilities and Restraints Catalog includes Part I, Mission Capabilities, which treats general mission performance aspects of the Agena vehicle with Atlas and Thrust Augmented Thor vehicles. This second volume, including Parts II through VII, presents in detail the hardware and operational aspects of the several vehicle combinations.

The purpose of this catalog is to establish meaningful general guidelines for the spacecraft designer or integration engineer interested in applying the Agena to a particular mission. In general, the philosophy guiding the selection and presentation of the available information has been to cite certain representative mission configurations and data. The Mariner Mars mission, the first NASA Agena D application, is particularly appropriate for this purpose. The user of this catalog is cautioned, however, that the data presented are for preliminary illustrative use only, and subject to modification for the particular mission.

Order of the presentation in Volume II has been arranged as follows:

- Part II, Launch Vehicle Familiarization, includes material relative to the overall launch vehicle coordinate system, and the Atlas and TAT booster vehicle descriptions
- Part III, Spacecraft Restraints, provides several sections on material of prime interest to the spacecraft designer: loads, dynamics, and clearances and tolerances, etc., on representative available hardware at the interface between the Agena and spacecraft
- Part IV, Spacecraft Support, includes a detail hardware description of available support items developed and qualified for prior missions
- Part V, Agena Vehicle Systems, describes the various basic Agena subsystems in detail, together with descriptions of optional or program peculiar equipment available for various mission applications

- <u>Part VI, Launch Operations</u>, reviews the AGE support hardware generally required in one section, and the other section fully describes the typical procedural routine attendant upon launches from the Eastern and Western Test Ranges with Atlas or TAT
- <u>Part VII, Programming</u>, provides generalized discussions relative to the typical Agena factory program, with particular emphasis on interface hardware requirements imposed by Agena upon the spacecraft program for tests, etc.

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SECTION 5
LAUNCH VEHICLE FAMILIARIZATION

5.1 DEFINITIONS AND COORDINATE SYSTEMS

The launch vehicle may be defined as the complete vehicle as launched in support of a given mission, and prior to any staging or other in-flight separation of components or systems. In respect to Agena applications, the launch vehicle consists of a spacecraft, Agena vehicle, and primary booster. To date, Agena vehicles have been utilized with Atlas D*, Standard Atlas (SLV-3), Thor (SLV-2 or DM-21)*, and Thrust Augmented Thor (LV-2A or TAT) boosters.

The spacecraft is usually a complete vehicle system designed to function as an orbit, lunar or interplanetary platform for scientific or experimental payloads. The spacecraft will not, in most cases, include primary propulsion rockets and the principal velocity increments (i.e., placement into earth orbit or interplanetary trajectory) will be the function of the booster/Agena stages. More than one spacecraft can be placed into orbit by a single launch vehicle, as in ISIS-X and some military programs. (Volume I has summarized the types of spacecraft injections which the launch vehicle has the capability of performing.)

The Agena is generally employed as an upper stage booster supplying energy by means of one or two engine burning periods to transfer the spacecraft from the primary booster trajectory to a specified earth orbit or escape trajectory. In NASA applications with separable spacecraft, the Agena portion of the mission usually ends shortly after separation. (Part IV of this catalog provides detailed descriptions of the Agena. Volume I, Part I, describes the manner of application.)

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*This catalog will be limited in most cases to information on the available launch vehicle systems. For future programs, only Atlas SLV-3 and TAT boosters, and Agena D (S-O1B, SS-O1B) vehicles will be employed.

The launch vehicle primary stage is the Atlas SLV-3 or TAT booster. Function of this stage is to provide sufficient energy to place the Agena/spacecraft into an 85 to 100 nm ballistic trajectory. (The primary booster vehicles are described in detail later in this section.)

5.1.1 Launch Vehicle Interface Definition

Interfaces may be defined as the physical junction and associated mechanical and electrical hardware between major launch vehicle systems as specified by associate contractor design responsibilities, and also all of the physical, functional and environmental characteristics of the launch vehicle that directly affect interface hardware design. Only the physical interfaces are described in this section. (Characteristics and parameters affecting intervehicle design are presented in Sections 6 through 9 under "Spacecraft Restraints.")

The Agena/booster interface consists of the hardware at the junction between the Agena and primary booster. This interface is basically a zone on both sides of the plane which separates the booster adapter from the primary booster. Significant components at the interface include the structural attachments and faying surfaces, electrical disconnect and any protrusion which extends into the interface zone between the Agena and booster. The Agena/booster interface is located at LMSC Station 526 (Atlas) and 492.2 (TAT) as shown in Figs. 5-1 and 5-2 respectively. Further details of the Agena/booster interface are given in Fig. 5-3.

The spacecraft/Agena interface consists of the hardware at the junction between the spacecraft and the spacecraft adapter plus shroud. In essence this interface is a zone on both sides of the plane which separates the spacecraft from the adapter, and an envelope surrounding the spacecraft but inside of the shroud.

Significant components in the interface include the spacecraft separation system, faying surfaces between spacecraft and adapter, spacecraft electrical disconnect, spacecraft umbilical connector, and any protrusion from the shroud or adapter that extends in proximity to the spacecraft.

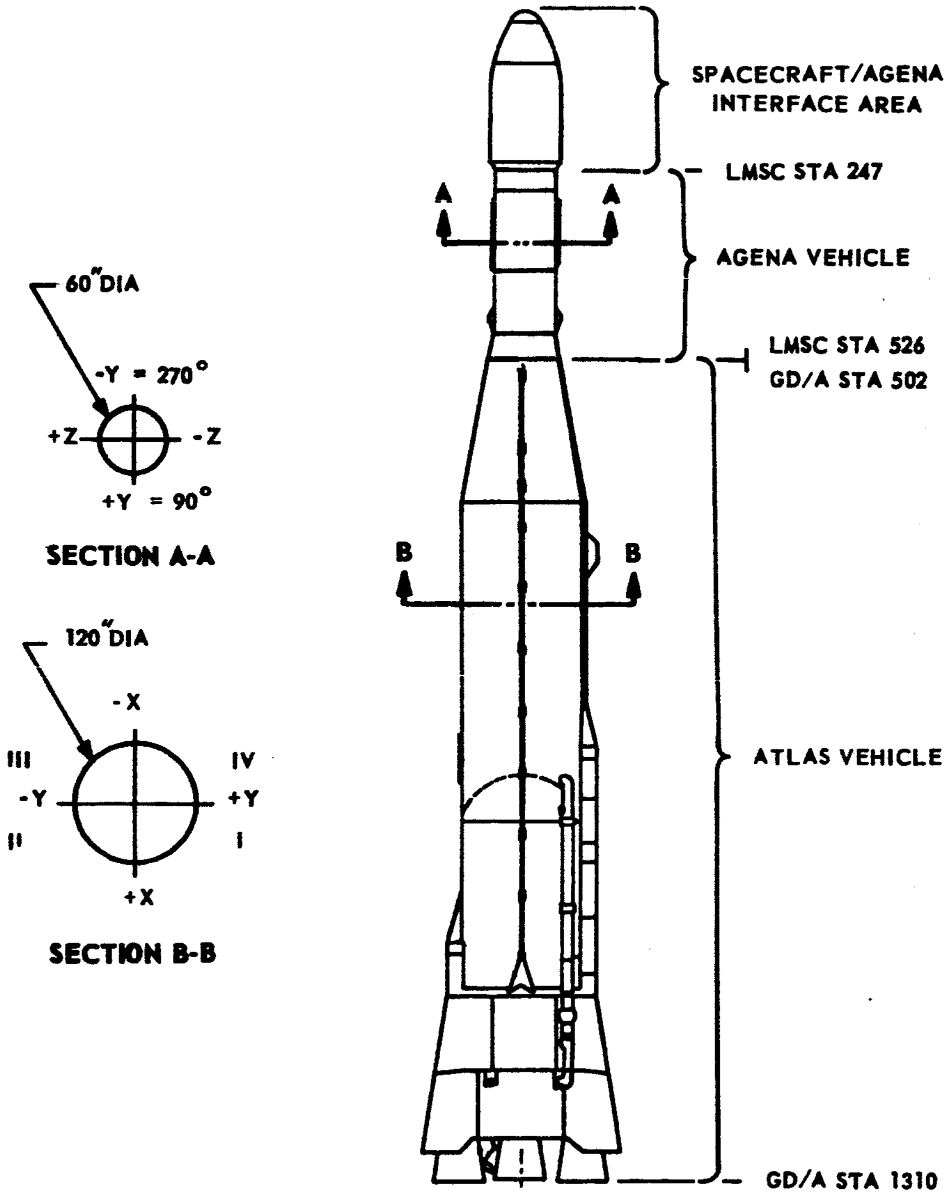

Figure 5-1 Standard Atlas/Agena D Launch Vehicle (Lunar Orbiter)

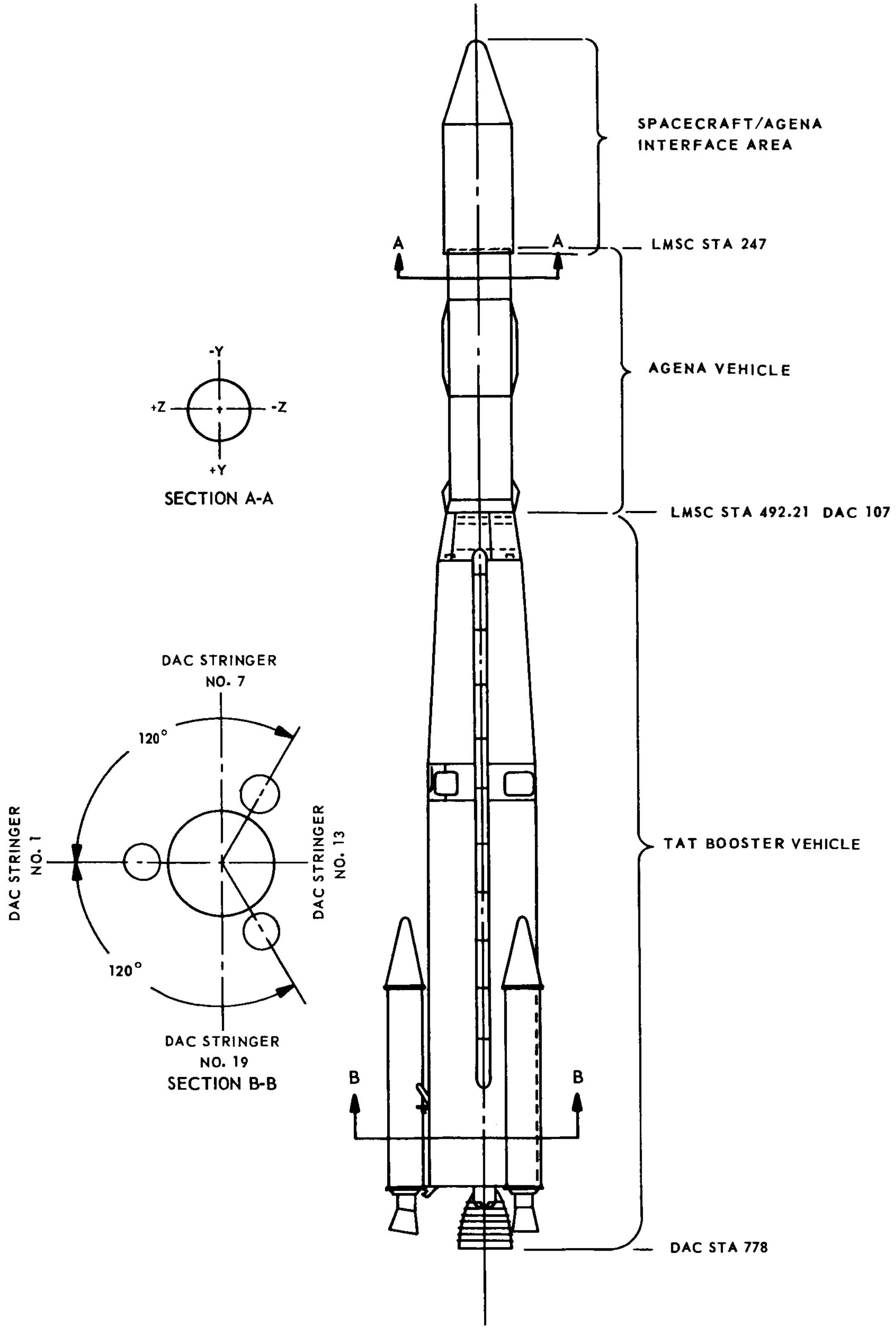

Figure 5-2 TAT/Agena D Launch Vehicle (POGO)

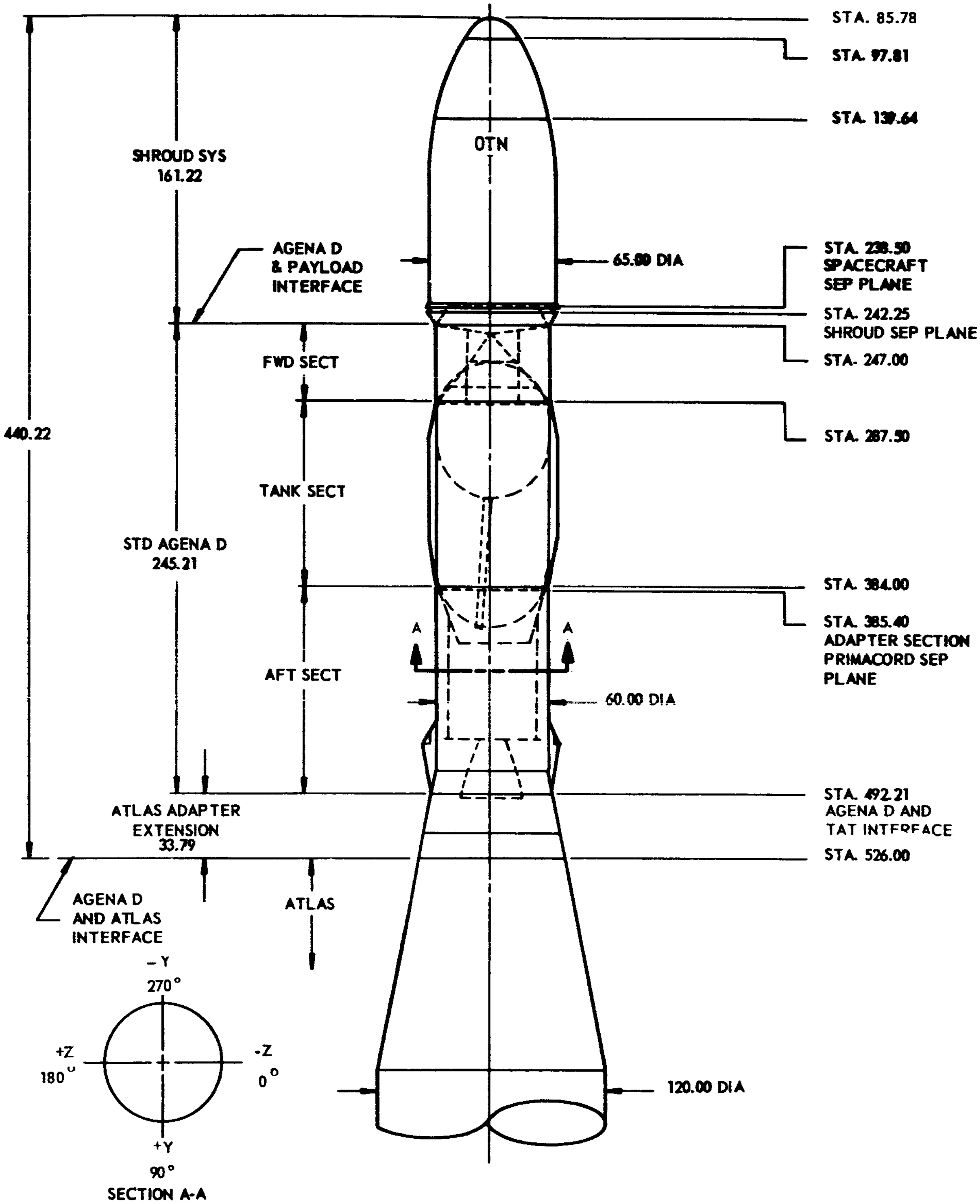

Fig. 5-3 Agena D Interface and Station Orientation (Lunar Orbiter)

LMSC station numbers of the spacecraft/Agena interface can vary depending on the length of spacecraft adapter and shroud configuration. The spacecraft/Agena interface is identified in Figs. 5-1 and 5-2 for Lunar Orbiter and POGO missions respectively.

The junction between the overall launch vehicle and its Aerospace Ground Equipment is also generally regarded as an interface. Interface hardware includes any AGE surfaces which contact the launch vehicle and umbilical connections for supplying fuel, power, signals,or air conditioning. Details of AGE/vehicle interfaces and coordinates are given in Section 18.

5.1.2 Launch Vehicle Coordinate Systems

Coordinate systems for the launch vehicle are specified early in each program to facilitate integration of the flight hardware. Coordinates are expressed in terms of station numbers and mutually perpendicular axes. Station numbers identify position in inches along the longitudinal axis of the launch vehicle. Axes are labeled X, Y, and Z with plus or minus signs assigned to directions along the axes. The axes perpendicular to the launch vehicle longitudinal axis divide the cross section into quadrants numbered clockwise when looking forward. Angular position is identified in degrees from the reference axis or in a particular quadrant. The axes are fixed with respect to the launch vehicle regardless of attitude or flight phase.

The coordinate systems for the Agena D, Atlas SLV-3 and TAT are standardized in relation to each other for all missions. Each spacecraft has its own individual coordinates and, therefore, the relative angular position between the the spacecraft and Agena/booster is usually different. The Spacecraft Center should indicate their coordinate system in relation to the Agena D system. The launch pad arrangement of the launch vehicle with respect to true North should also be specified as part of the coordinates. Coordinate systems for Lunar Orbiter and POGO launch vehicles are given in Figs. 5-1 and 5-2 to indicate typical examples for those vehicles employing Atlas and TAT boosters.

5.1.3 Agena Coordinates

Agena D station numbers increase in the aft direction from an arbitrary reference point forward of the vehicle designated LMSC Station 0 to Station 526.00 (Fig. 5-3). Vehicle movement about the axes are termed as pitch, roll, and yaw. Pitch movement of the vehicle results in rotation about the Y axis. Yaw movement of the vehicle results in rotation about the Z axis. These motions are shown in Fig. 5-4. The angular reference system provides a clockwise 360-degree orientation about the longitudinal or X axis of the vehicle. The angular reference begins with 0 degrees at the -Z axis end, and continues in a clockwise direction (looking forward) through 90 degrees at the +Y axis, 180 degrees at the +Z axis, 270 degrees at the -Y axis, and back to the -Z axis for 360 degrees.

5.1.4 Launch Facilities

Facilities for TAT/Agena launches are available only at the Western Test Range (WTR). Facilities for Atlas/Agena launches exist at both Eastern and Western Test Ranges. (Section 18 delineates the AGE and facilities currently available for these launch vehicle configurations.)

5.2 AGENA VEHICLE SUMMARY

The Agena D is a standardized vehicle that can be configured to perform as an intermediate stage booster or as an orbital vehicle. Important features include: a standardized spacecraft support system interface, an integrated guidance module, an integrated power system, a standardized booster interface and, wherever possible, maximum component accessibility. There are essentially five subsystem (SS) categories of equipment that make up the Agena vehicle. These are Spaceframe (SS/A); Propulsion (SS/B); Electrical (SS/C); Guidance and Control (SS/D), Communications and Control (SS/C&C). These subsystems are discussed in detail in Sections 13 through 17 respectively. Additional Agena D information for the spacecraft contractor is presented in the following paragraphs.

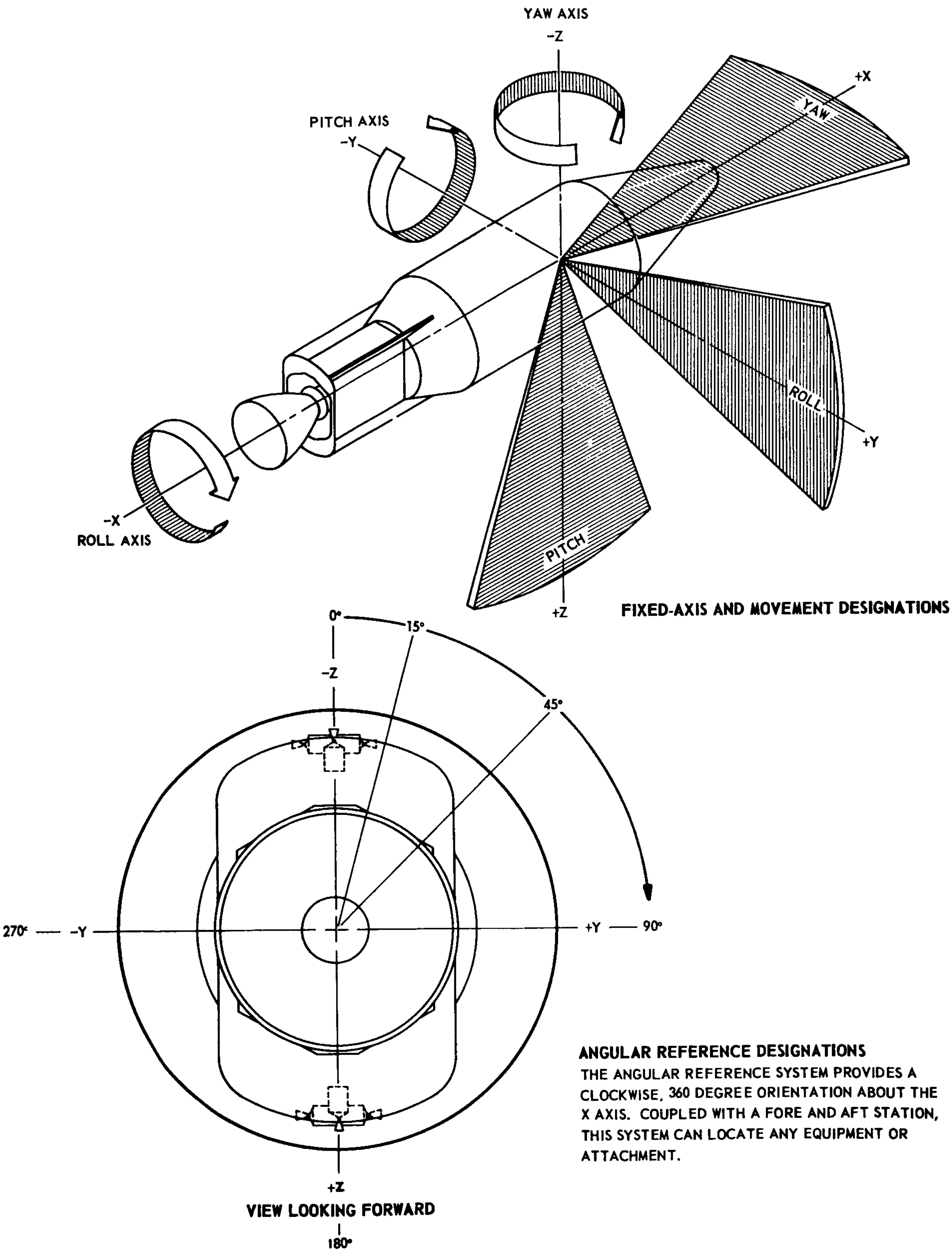

Figure 5-4 Agena D Fixed-Axis and Movement Designations

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5.2.1 Equipment Definitions

To meet the requirements of performing various mission functions, the Agena vehicle utilizes three categories of equipment in each of the subsystems discussed above. These categories are basic equipment, optional equipment, and program or mission peculiar equipment. The basic vehicle consists of structure and equipment common to most of the using space programs.

Additional elements required for flight capability are selected from a group of Agena D qualified optional flight items. Mission capability is completed by the installation of a second group of items identified as program or mission peculiar. These equipment definitions are expanded in the Glossary.

5.2.2 Vehicle Assembly Procedures

The basic Agena D is assembled into the defined configuration (AD 68 and up) and tested by LMSC's Agena D manufacturing organization under Air Force contract. After tests as a complete system, the vehicle is accepted by the Air Force Satellite Systems Division (AFSSD) in an initial DD250 procedure. The basic vehicle is then assigned by AFSSD to using programs as Government Furnished Equipment (GFE). In NASA Agena applications the vehicle, under NASA direction, is modified by the addition of optional and program peculiar equipments, which are installed in the basic Agena D vehicle to adapt it to a given mission. In certain cases, selected items may be removed from the basic vehicle; these are identified as "permissible removal" in the basic vehicle master breakdown. (A description of the typical NASA Agena program will be found in Section 20.)

5.3 STANDARD ATLAS BOOSTER (SLV-3)

The standardized Atlas launch vehicle developed for space missions is a modified SM-65D (LV-3A), and is designated the SLV-3. The improvements in the vehicle are such that the basic vehicle is adaptable to specific mission requirements by means of mission-peculiar kits.

Standardization has resulted in several changes from the SM-65D missile previously used as a space launch vehicle. The B-1 equipment pod (Fig. 5-5), containing the basic electrical and instrumentation wiring utilized for all missions, allows for installation of a program peculiar electrical distribution box kit for adapting the standard vehicle to a particular mission and eliminating wiring splices. Similarly, in the B-2 pod, the basic vehicle allows interchangeable installation of the two GE guidance kits, the Mod III-G used at ETR or the Mark II used at WTR. Other features of the standardized Atlas, as compared with the Atlas D, include:

- Higher tank pressurization and heavier skin gauges
- Propellant utilization system using liquid level sensing as provided for Atlas Series E and F missiles
- Derated verniers, from 1000 lb to 670 lb at sea level
- Space for additional TLM package
- Uprated booster engines, from 154, 500 lb to 165, 000 lb
- Provision for mission peculiar variations in the autopilot

5.3.1 Airframe

The Atlas airframe consists of three sections: the nose, tank, and thrust sections. A distinguishing feature of the Atlas structure is the extensive use of welded stainless steel sheet (0.05 to 0.048 in thickness) particularly in the tank section, which is a thin wall, fully monocoque structure pressure vessel, deriving its rigidity and strength from internal pressurization. Tank diameter is 120 in., tapering at the forward end to a 69.8 in. hemisphere, and terminating in a 90 degree cone at the aft end. Total length of the tank section is approximately 50 ft. An ellipsoidal diaphragm divides the tank into two compartments: a forward tank section of 2, 503 cu. ft. for liquid oxygen, and an aft tank section of 1, 557 cu. ft. for RP-1 (hydrocarbon) fuel. A thrust ring with a webbed bulkhead at the aft end prevents radial distortion of the tank from stresses created by engine thrust. The liquid oxygen engine feed line, tank pressurization lines, and cabling are attached

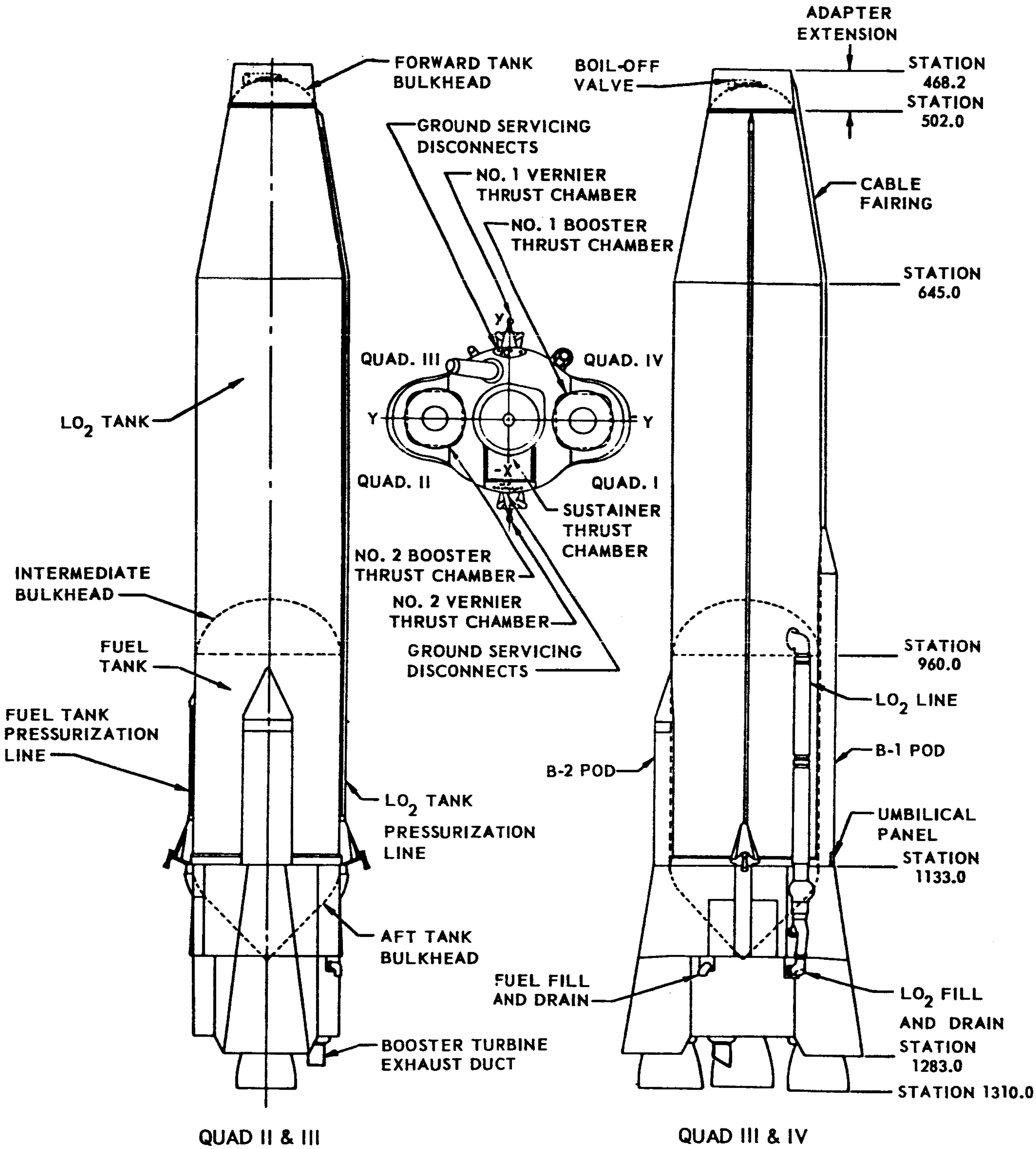

Figure 5-5 Atlas Configuration

external to the tanks, as are the equipment pods containing the electrical and electronic units. The aft section consists of a structure housing the two booster engines and their associated equipment. The structure consists of a thrust cylinder, heat shield, and engine nacelles or skirts that form a single compartment. Attachment to the tank section is at the thrust ring near the aft end of the tank, with provision for separation in flight.

5.3.2 Propulsion

The Atlas propulsion system consists of three main liquid propellant (liquid oxygen and RP-1 hydrocarbon) rocket engine assemblies: two turbopump-fed YLR89-NA-7 boosters, turbopump-fed YLR105-NA-7 sustainer, and two pressure-fed YLR101-NA-15 verniers, all manufactured by Rocketdyne Division of NAA, and designated as the MA-5 group. These engines are single-start with a constant propellant flow rate. Operation may be characterized as the "stage and a half" type with all engines ignited prior to liftoff, booster burnout and staging at approximately T + 135, followed by a continuing sustainer burn period of some 190 seconds, and terminating in a vernier-only operation for the balance of the period of the booster-powered flight.

5.3.2.1 Booster Engine. The booster engine system (Fig. 5-6) consists of a dual turbopump power package and two thrust chambers. The two turbopumps are dual centrifugal units for fuel and oxidizer and are both driven by a single geared turbine actuated by hot gas from a gas generator. The two thrust chambers and nozzles are regeneratively cooled by the fuel supply to each chamber. The thrust chamber and nozzles are independently gimbaled in pitch and yaw by means of hydraulic actuators, and have a maximum deflection of 5 degrees about the longitudinal axis of the vehicle. Control of the vehicle attitude in pitch, yaw, and roll during the booster burn period is controlled by means of nozzle movements commanded by the flight control (autopilot) system. (Roll control by the booster engine system is exercised by independent control of the two nozzles.)

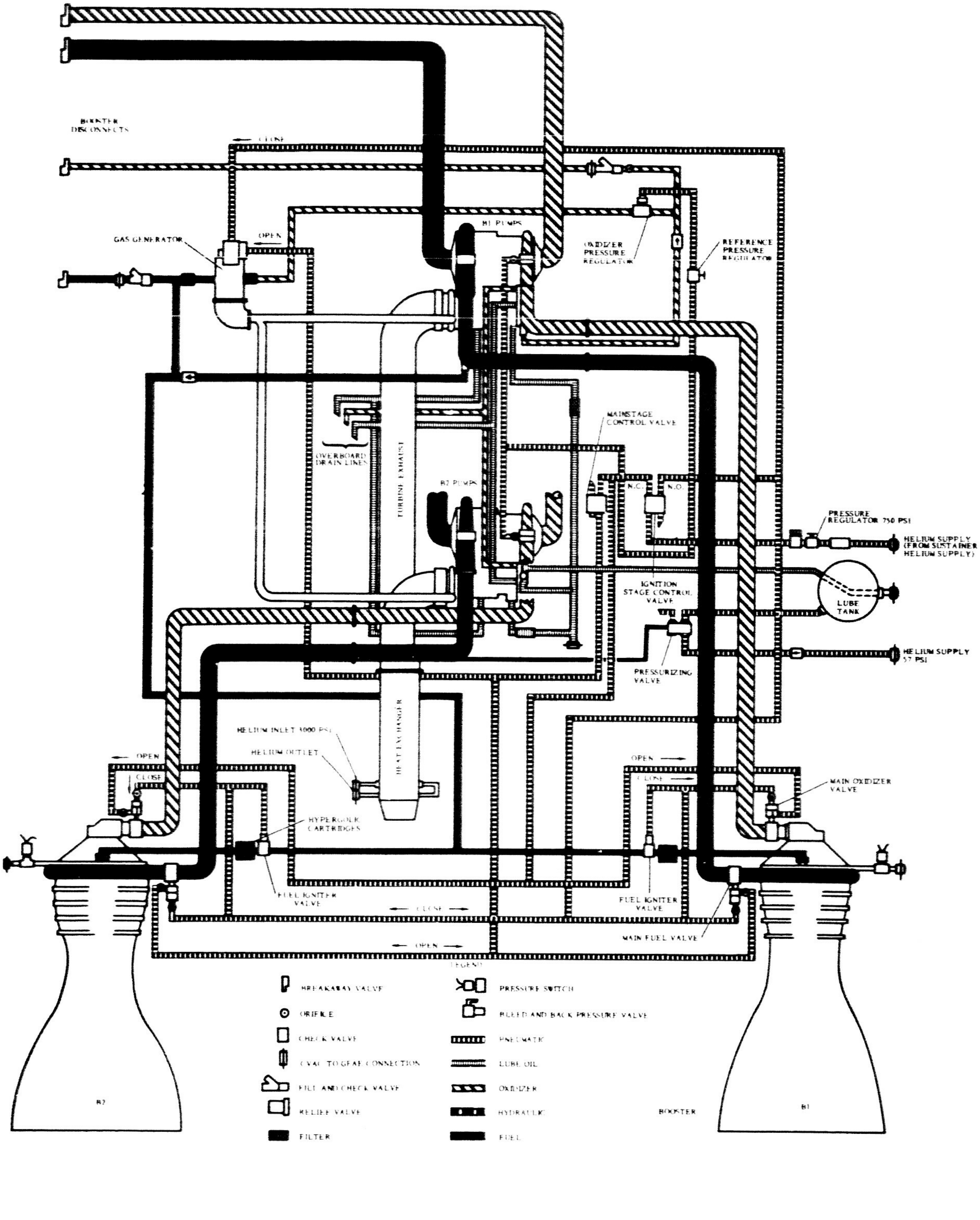

Figure 5-6 Atlas Booster Engine System

5.3.2.2 Sustainer Engine. The Atlas sustainer engine system (Fig. 5-7) is a dual turbopump and integral single-nozzle unit located between the booster nozzles. The general features and operation are similar to that of the booster engine system. This engine is gimbaled to 3 degrees to provide attitude control in pitch and yaw during the sustainer flight phase after BECO. (The sustainer engine is not gimbaled during booster phase flight.)

5.3.2.3 Vernier Engines. The two vernier engines are separate propulsion units mounted on the airframe 180 degrees apart at the base of the fuel tanks. The verniers operate throughout powered flight. Fuel and liquid oxygen propellants are supplied from vernier start tanks during starting (pre-liftoff) and vernier solo (post-SECO) periods; and, while the sustainer is operating, by the sustainer turbopump. Each vernier thrust chamber is regeneratively cooled by fuel. The hydraulic gimbal mechanism permits pitch and roll movement through an arc 20 degrees inboard and 30 degrees outboard from a zero position canted 45 degrees outboard from the longitudinal axis. During boost phase flight the vernier engines provide for small vehicle attitude adjustments in the roll axis; during sustainer operation they provide the full control in roll; and during vernier solo they provide attitude control about all axes.

5.3.3 Guidance and Flight Control

The directional and velocity control of the Atlas is achieved by similar means in both ETR and WTR launches. The Atlas guidance consists of two related subsystems: the flight control (autopilot) and the GE radio guidance. With respect to the four main phases of the Atlas flight — booster phase, staging, sustainer phase, and vernier solo — the customary flight uses the programmed autopilot in an "open loop" in the booster phase. BECO and the subsequent booster staging events are normally initiated by radio guidance command, as are SECO, VECO, and Agena staging. Radio guidance steering commands,

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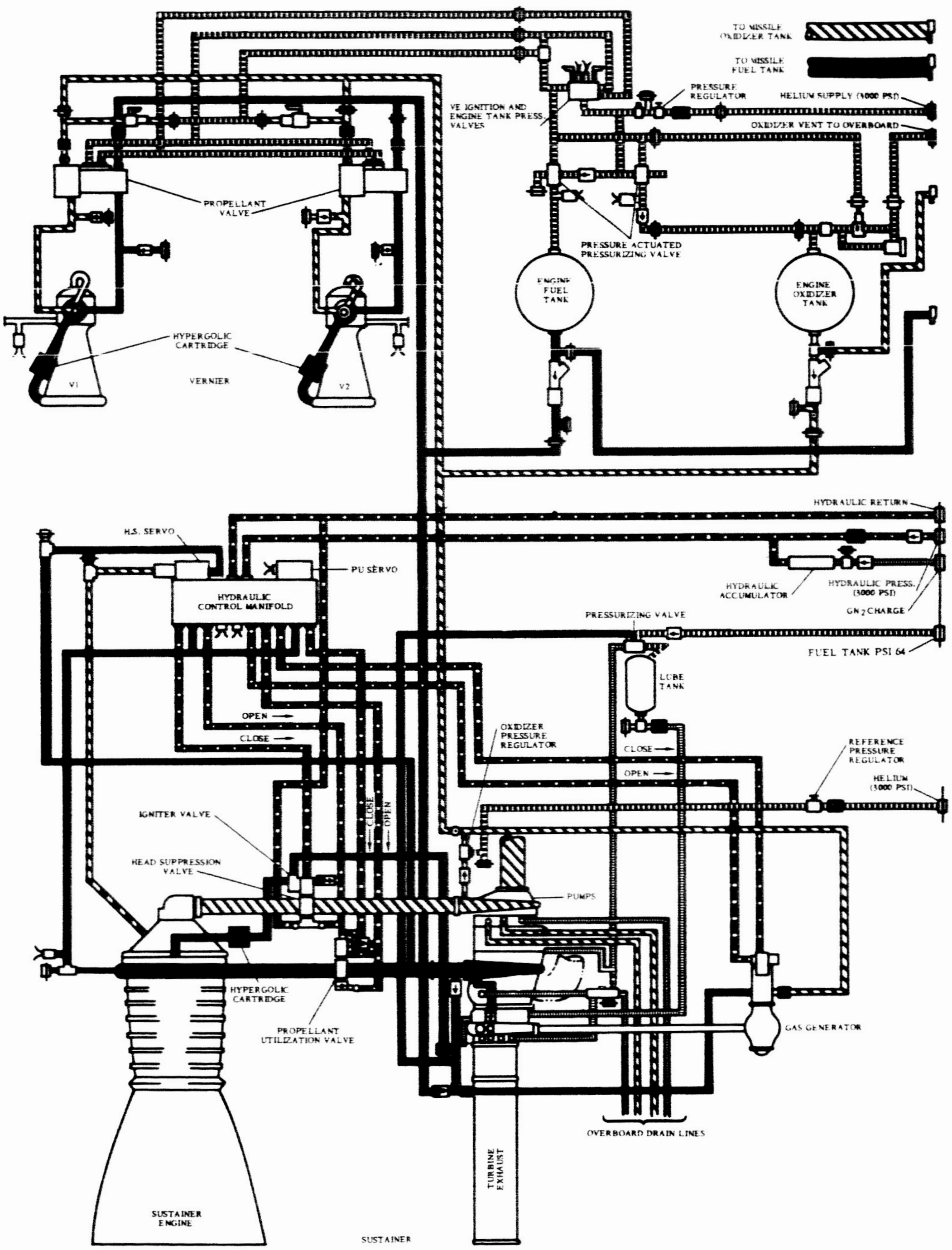

Figure 5-7 Atlas Sustainer and Vernier Engine Systems

based on radar tracking information, are utilized in the sustainer phase to correct deviations from the desired trajectory*.

5.3.3.1 Flight Control Subsystem (Autopilot). The Atlas flight control subsystem (Fig. 5-8) consists basically of the flight programmer, gyro reference package, the servo control electronics, and hydraulic controllers. Timing and switching functions are performed by the flight programmer utilizing command inputs from the radio guidance for certain events.

Steering commands from the flight programmer or radio guidance are sent to the gyro package which monitors the instantaneous difference between actual and desired vehicle attitude. In each of the three axes – pitch, roll, and yaw – single-degree-of-freedom, rate-integrating displacement gyros form a prime reference, and steering commands are effected by torquing these gyros. Signals proportional to the difference between actual and desired vehicle attitude, as measured by the gimbal angle, are sent to the respective gyro signal amplifier for input to the servo control units. Rate damping is provided by signals generated by two rate gyros which sense vehicle angular rates in pitch and yaw and introduce corrective signals into the gyro signal amplifiers.

5.3.3.2 Radio Guidance Subsystem. The Atlas guidance subsystem consists of a ground and vehicle systems as illustrated in block diagram form in Fig. 5-9.

The main ground elements are the monopulse X-band position radar, continuous-wave L-band doppler radar system to measure velocity, and a Burroughs computer. Airborne components include the GE guidance package, with rate beacon, pulse command beacon and decoder. The Atlas autopilot is tied with this system, accepting decoded commands via the pulse beacon and position radar.

*Booster phase radio guidance steering may, however, be utilized to increase accuracy and total ascent performance for certain missions.

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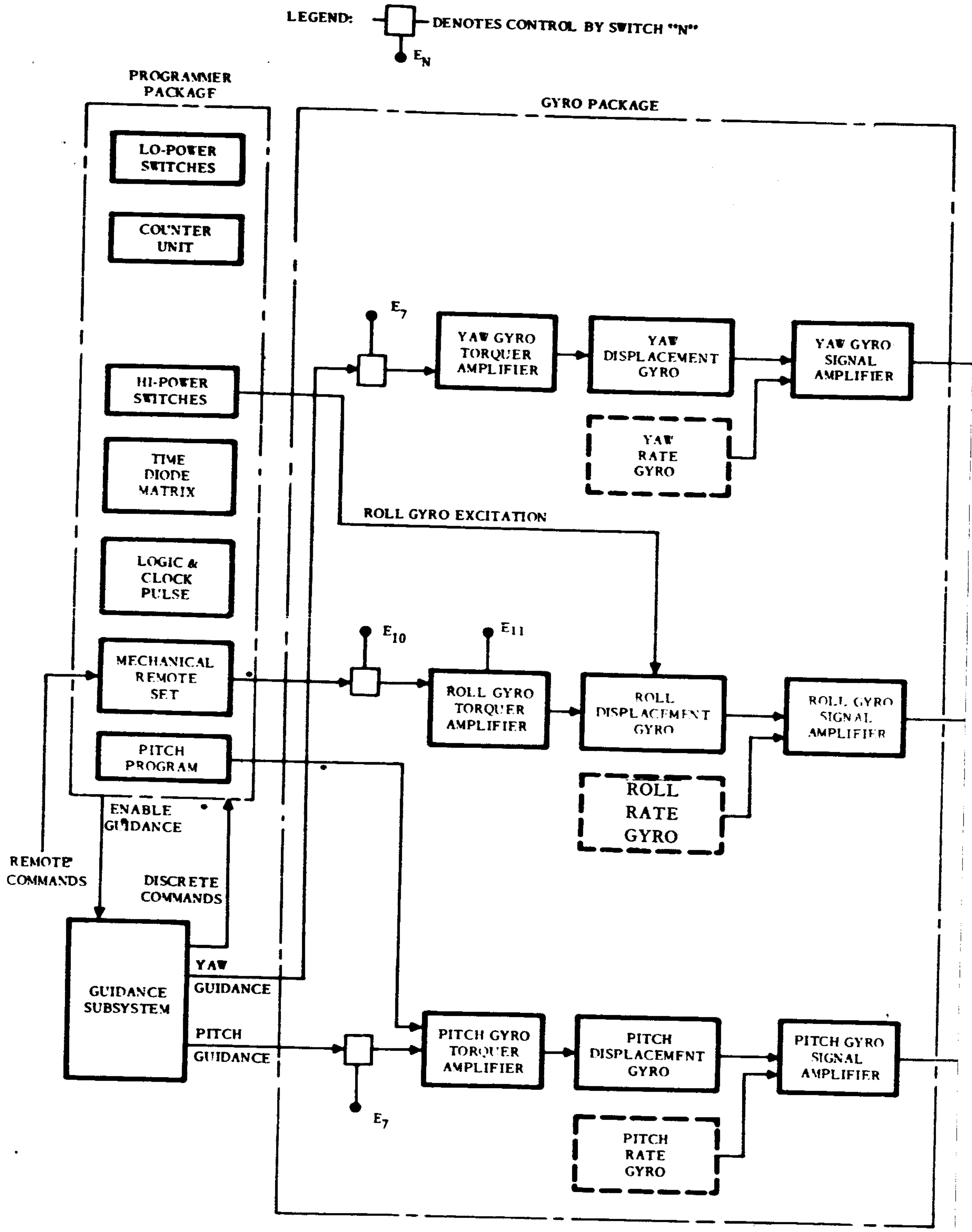

LEGEND: DENOTES CONTROL BY SWITCH "N"
E_N
PROGRAMMER PACKAGE
LO-POWER SWITCHES
COUNTER UNIT
HI-POWER SWITCHES
TIME DIODE MATRIX
LOGIC & CLOCK PULSE
MECHANICAL REMOTE SET
PITCH PROGRAM
GYRO PACKAGE
E_7
YAW GYRO TORQUER AMPLIFIER
YAW DISPLACEMENT GYRO
YAW GYRO SIGNAL AMPLIFIER
YAW RATE GYRO
ROLL GYRO EXCITATION
E_{10}
E_{11}
ROLL GYRO TORQUER AMPLIFIER
ROLL DISPLACEMENT GYRO
ROLL GYRO SIGNAL AMPLIFIER
ROLL RATE GYRO
ENABLE GUIDANCE
REMOTE COMMANDS
DISCRETE COMMANDS
GUIDANCE SUBSYSTEM
YAW GUIDANCE
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PITCH GYRO SIGNAL AMPLIFIER
E_7
PITCH RATE GYRO

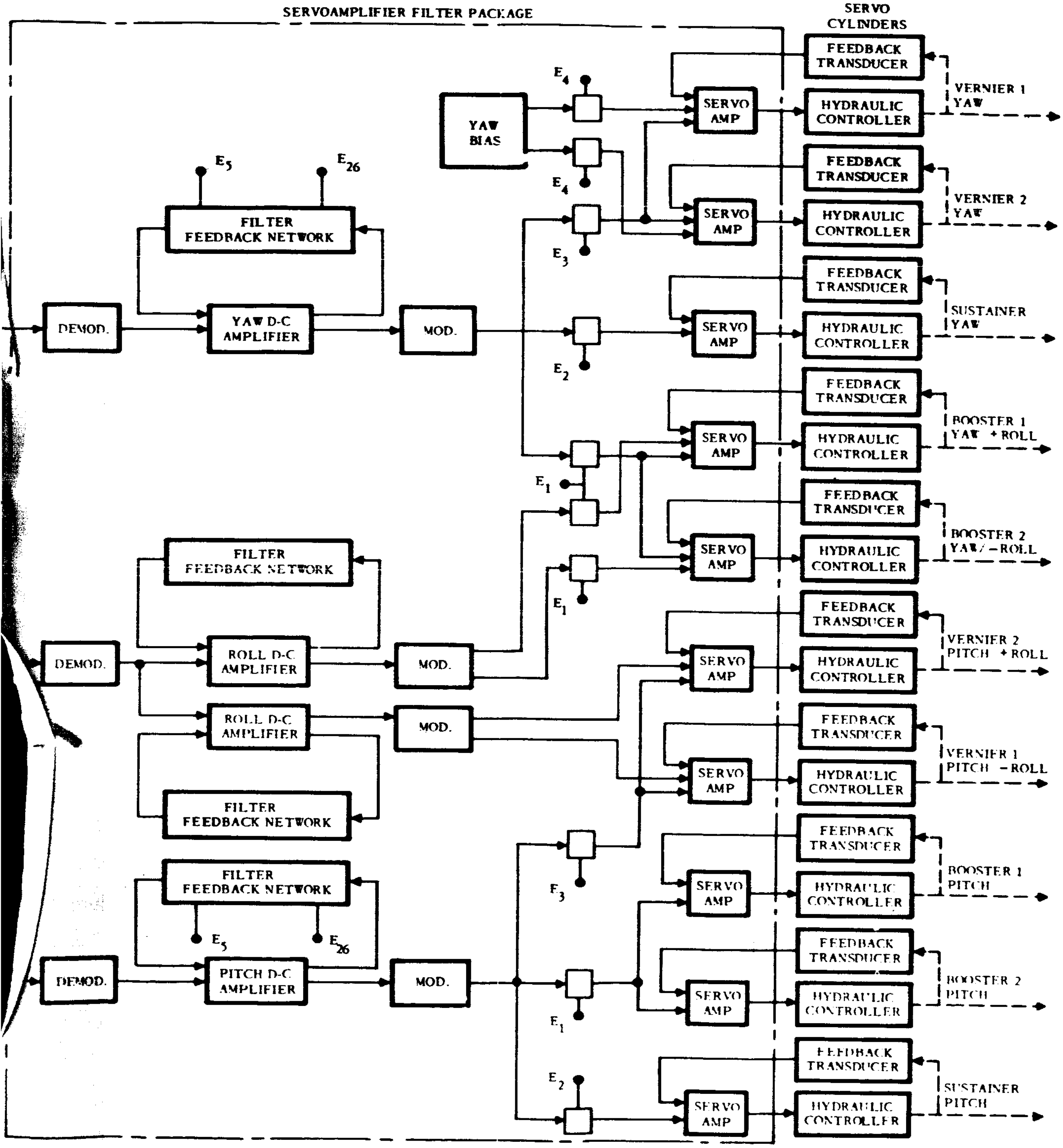

Figure 5-8 Atlas Flight Control Subsystem

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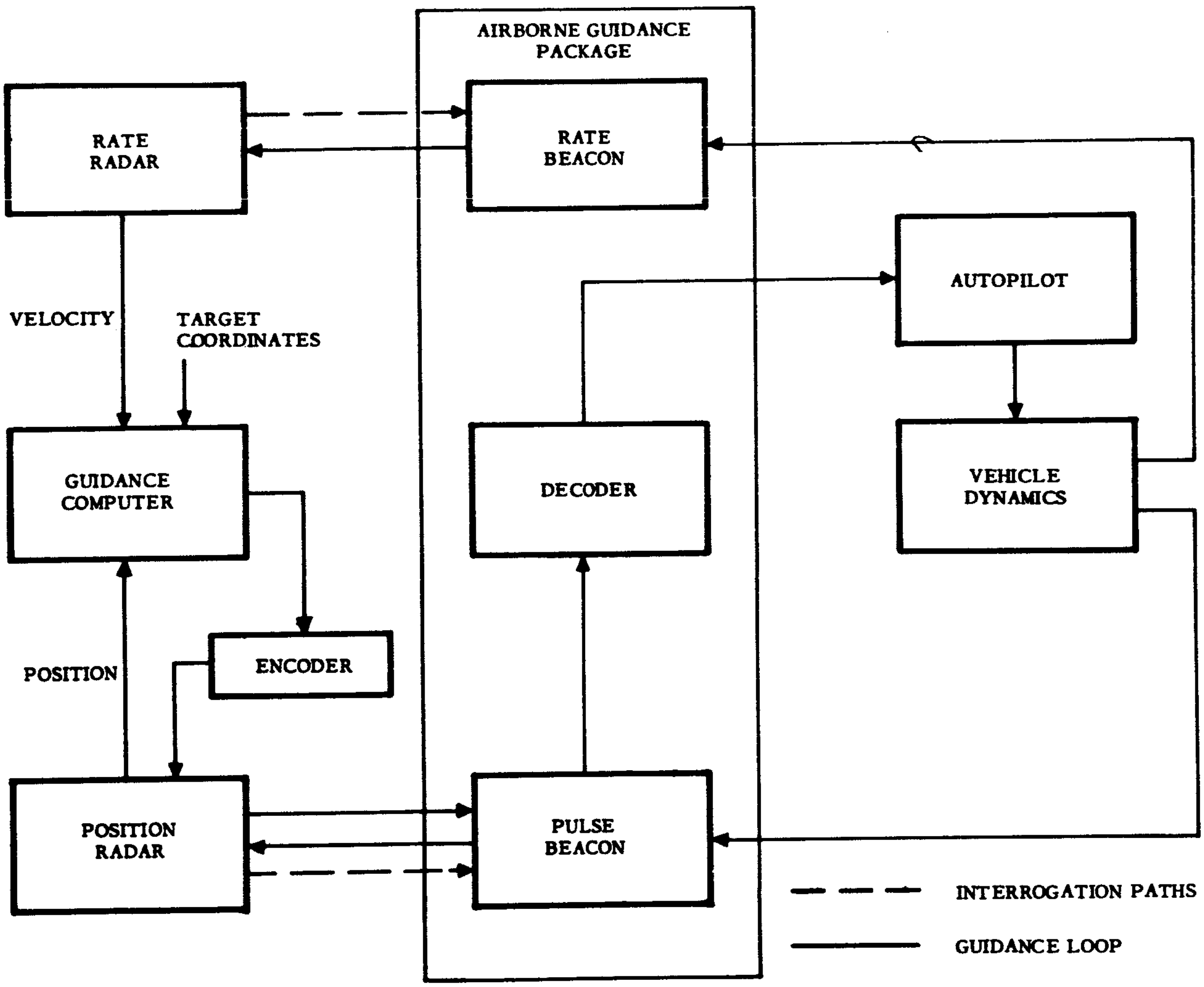

Figure 5-9 Atlas Guidance Loop

The radio inertial guidance system is usually enabled (command capability) shortly after booster staging, and operates until sustainer thrust termination. Prior to enabling, the Atlas flies open loop by the preprogrammed autopilot.

Logic functions are performed by the ground-based Burroughs computer, using a set of explicit guidance equations representing the desired radius, inertial velocity and flight path angle conditions at the termination of Atlas thrust. The Agena gyro orientation, velocity, and time references are established by the conditions prevailing at Atlas VECO, when the Agena gyros are uncaged and the Agena timers are initiated by a guidance discrete before separation.

5.3.4 Other Atlas Airborne Systems

Other major Atlas subsystems are:

a. Pneumatic — provides helium at regulated pressures for propellant and turbopump lubrication tank pressurization, hydraulic reservoir pressurization, booster stage separation latch release, and main and vernier engine controls.

b. Hydraulic — powers the engine actuator system.

c. Propellant Utilization — provides a means of controlling propellant flow with greater accuracy than fixed metering orifices.

d. Electrical — providing battery supplied 28v dc and 115v ac, three phase, 400 cps power in flight together with provision for using ground base supplies prior to flight.

e. Telemeter — for instrumentation data on physical variables and component operation.

f. Range Safety and Flight Termination — separate units are used at ETR and WTR, and which provide means of terminating the flight by ground-commanded engine shutdown or explosive destruct.

5.4 THRUST AUGMENTED THOR (LV-2A)

The standardized TAT launch vehicle developed for space vehicle booster missions is a modified Thor DM-21, and is designated LV-2A. The main improvement in the TAT is the addition of three TX33-52 solid rocket booster motors (Fig. 5-10). Liftoff takes place with all motors burning, and three solid motors are jettisoned after burnout. Another feature of the TAT vehicle, as compared with the DM-21 includes relocation of the Bell Telephone Laboratory (BTL) radio guidance into the Agena.

5.4.1 Airframe

The main TAT airframe (exclusive of the boosters) consists of five sections: the forward (interstage) transition section, the fuel tank, the center body section between the propellant tanks, oxidizer tank, and engine section. Structures are principally fabricated of aluminum in a semimonocoque design with integral propellant tanks. The vehicle is self-supporting without pressurization of the propellant tanks. Flight control, auxiliary power, and other accessory subsystem assemblies are housed within the transition, center body, and engine sections. Two external longitudinal fairings provide for connecting cabling and tubing between these equipment sections. Total length of the TAT structure, from the forward interface ring to the aft ends of the launch support beams is approximately 51 ft. Vehicle diameter varies between 63.6 in. at the forward interface ring and 96 in. at the midsection. The midsection, oxidizer tank section, and engine sections have a constant diameter of 96 in.

The forward transition section is a truncated cone of conventional semimonocoque construction with rings and longitudinal stringers supporting the skin and attached to the forward end of the fuel tank. The section is 44 in. long. The forward ring forms an interface with the Agena booster adapter, which is bolted to the TAT and remains with the booster after inflight separation. Flush doors in the transition section provide access to the TAT electrical and flight control equipment installed in this section of the booster. An umbilical connection is also provided.

The TAT fuel tank structure is a truncated cone section of 0.25 in. aluminum sheet chemically milled in a waffle-like pattern on the inside surface, and with intermediate frames for added strength and rigidity. The structure terminates in dome shaped bulkheads fore and aft. Volume is approximately 643 cu. ft.

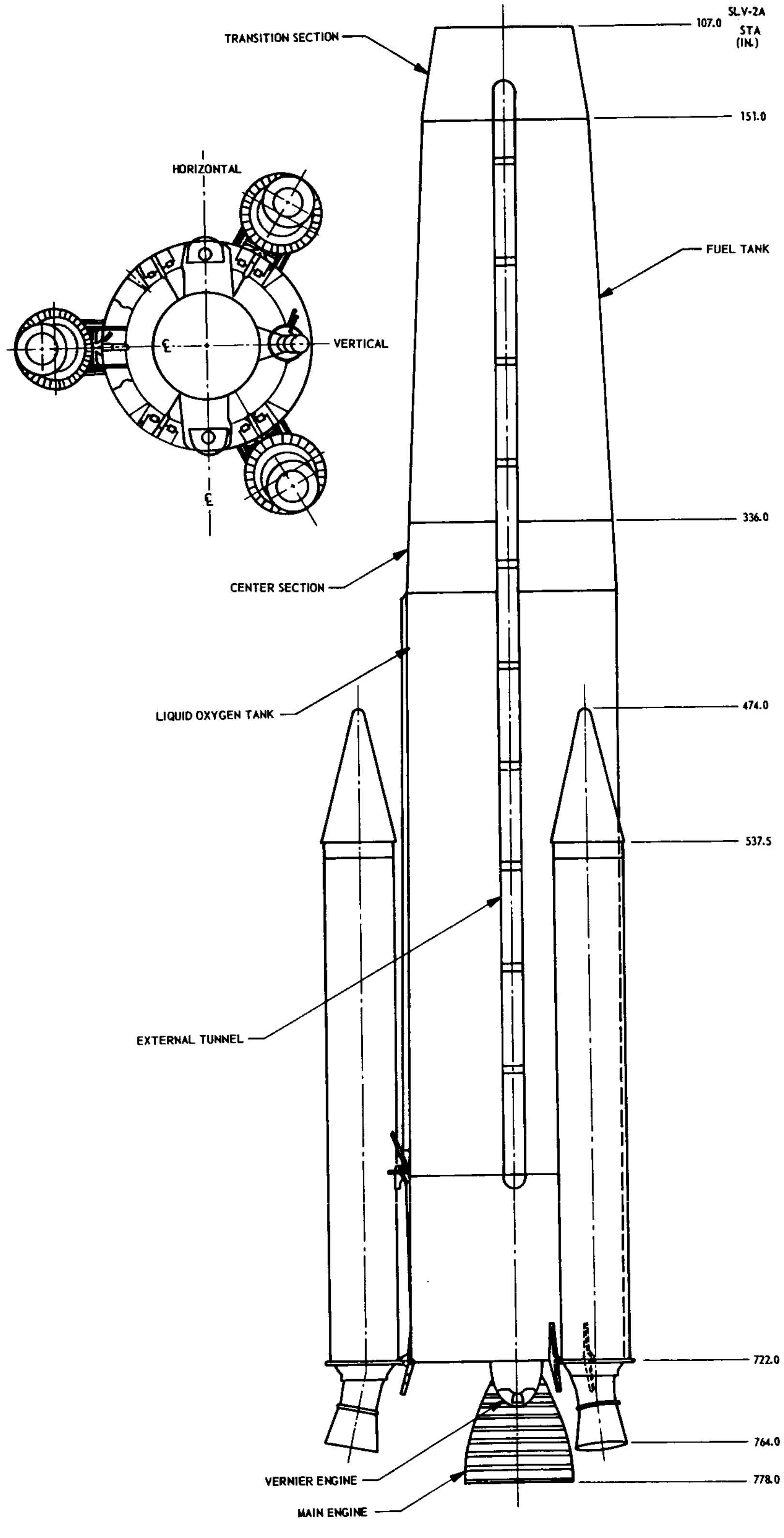

Figure 5-10 TAT Configuration

The center body section joining the two tank sections is a semi-monocoque cylinder with aluminum stringers, formers and skin. Two hinged doors provide access to the equipment and components mounted between the tanks.

The liquid oxygen (LOX) tank structure is similar to that of the fuel tank, but with a constant 96-inch cross-section. Insulating blankets are used over the dome-shaped bulkheads fore and aft to protect nearby equipment in the transition and engine sections from the extreme cold temperatures. A 36-inch skirt section extending aft from the LOX tank houses nitrogen gas supplies. A tunnel for the fuel supply line to the engine section runs through the LOX tank. Tank volume is approximately 998 cu. ft.

The engine and accessories section is generally of semi-monocoque design with aluminum stringers, formers,and skin. Three longitudinal structural beams 120 degrees apart transmit main engine thrust loads to the airframe. Interspaced between the thrust beams are three longitudinal launch beams in the aft third of the section, which transmit weight load of the vehicle to the launch stand when in the erected position. An end bulkhead with a 39-inch circular opening for the main engine thrust chamber also supports the vernier engines and encloses the engine compartment. (The opening for the engine thrust chamber is closed by a silicone rubber impregnated fiberglass curtain to prevent hot gas entry.) The main engine support fixture attaches to the thrust beams described, and the two verniers mount on the aft bulkhead with local structure protected from heating effects by fiberglass fairings or shields.

Structure associated with the three solid booster rocket motors includes the attach fittings and jettison mechanisms. The forward attachment point for each rocket motor is a socket joint on the forward part of an external longitudinal bracket attached directly to the TAT engine section. This socket transmits the major portion of thrust loads to the airframe. Sway braces and ejection links provide for positional stability and a positive ejection (radially outboard)when the socket joint is released by pyrotechnic bolt firing. A track and roller attachment at the aft end of the bracket supports the booster motor case in a manner which accommodates motor case expansion and contraction in flight and guides the case away from the vehicle during separation.

5.4.2 Propulsion

The propulsion system of the TAT consists of a main turbopump-fed liquid propellant engine, the two liquid propellant verniers and three jettisonable solid rocket motors. Liquid propellants are RJ-1 hydrocarbon fuel with liquid oxygen.

The main engine is the Rocketdyne MB-3 Block III, YLR-79-NA-13, which consists of two turbine-driven centrifugal pumps, a gas generator, and a gimbaled thrust chamber (see Fig. 5-1). The thrust chamber and nozzle are regeneratively cooled by fuel supply to the injector. Control of the vehicle in pitch and yaw is provided by nozzle gimbaling to 7 degrees in response to signals from the flight controller. Gimbaling is accomplished by hydraulic actuators.

The two vernier engines (Fig. 5-12) are Rocketdyne MB-3 Block II, XLR 101-NA-11, which are gimbaled to provide attitude control in pitch, roll, and yaw. Thrust of each is nominally 1150 lb at sea level. Nozzles are canted 6 degrees outboard from the vehicle center line and are capable of actuation 34 degrees in yaw, 45 degrees in pitch and 45 degrees (differentially, with respect to the other vernier) in roll.

The three solid rocket boosters are Thiokol TX33-52 motors with 55,850 lb thrust each at sea level. Burn time is approximately 36 seconds. Primary components of each motor includes a steel motor case containing the grain; a fixed position nozzle, canted 11 degrees outboard; and a DAC-attached aerodynamic cone fairing on the nose. Burning is initiated by electrical signal from TAT main engine fuel line pressure switch during

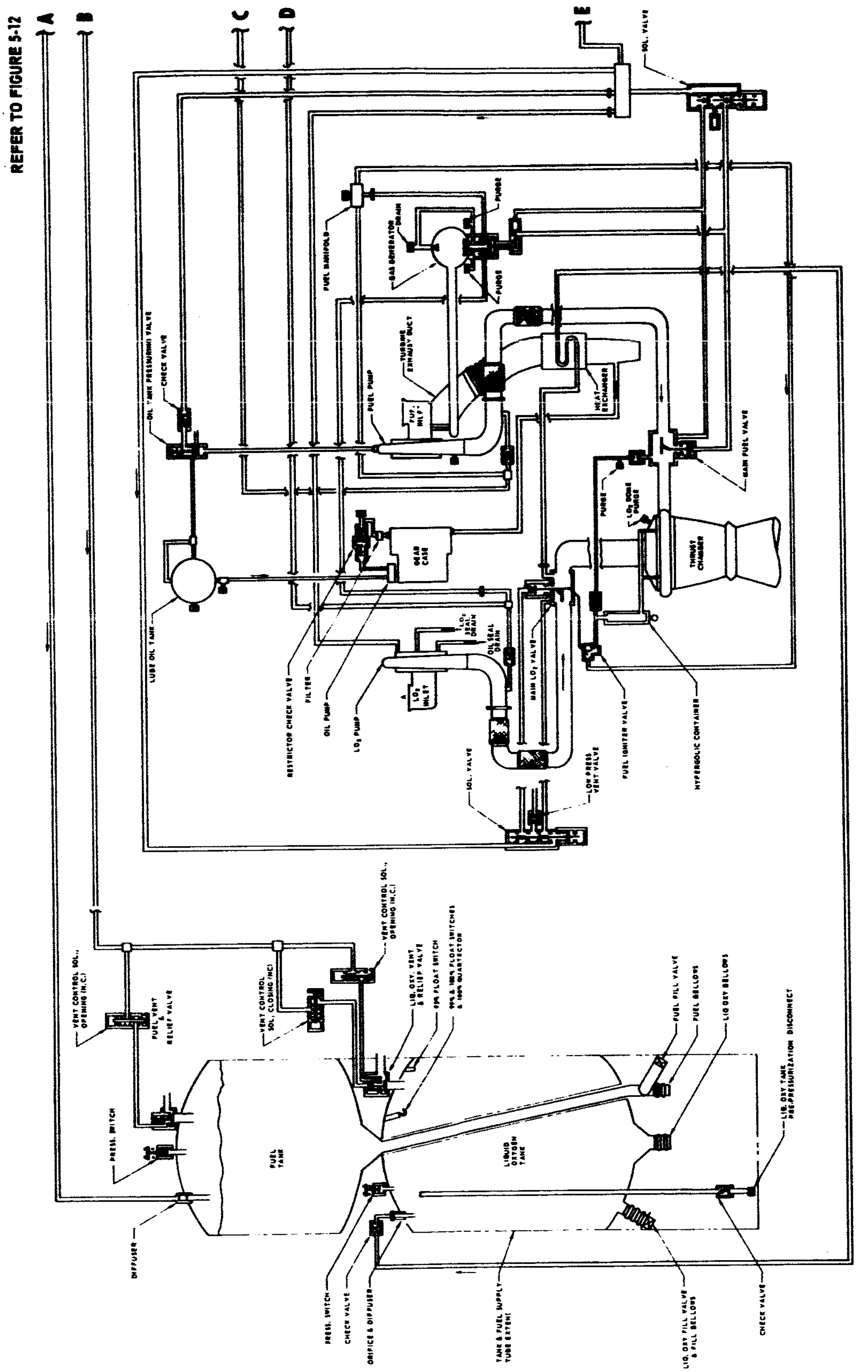

Figure 5-11 TAT Propulsion System (Sheet 1)

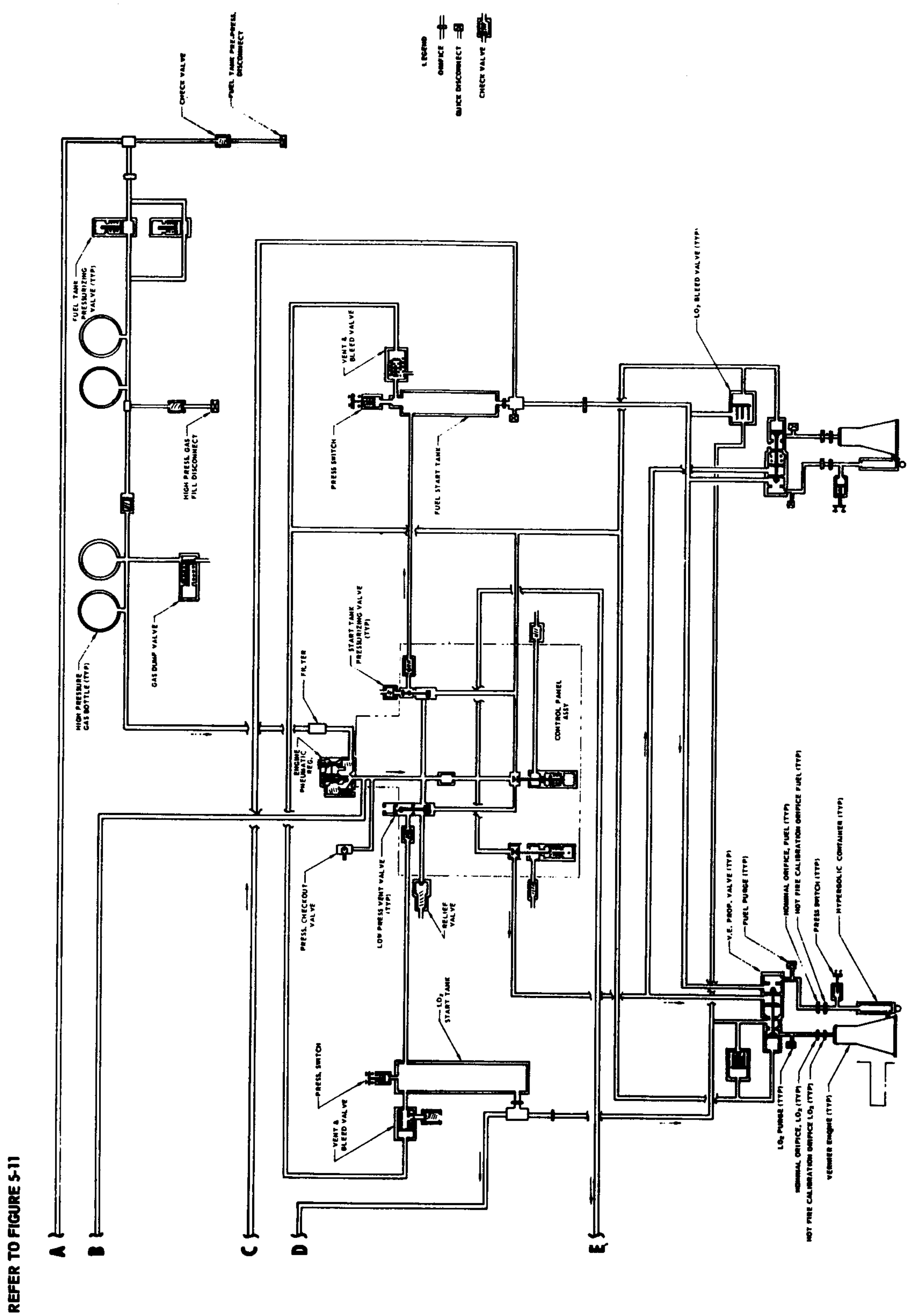

Figure 5-12 TAT Propulsion System (Sheet 2)

the main stage sequence. The motors are jettisoned by TAT programmer signal at approximately 65 seconds after ignition. Pyrotechnic bolt release of the socket clamp arrangement at the forward end of each rocket permits aerodynamic and continuing TAT acceleration forces at about 2.2g to deploy the spent motor cases radially outboard and away from the booster vehicle.

5.4.3 Guidance and Flight Control

The TAT guidance system consists of two related subsystems: the flight control (autopilot) and the BTL radio guidance. With respect to the normal ascent sequence the guidance portion is enabled at 90 sec, and provides the capability of ground commanded changes to correct for trajectory deviations. The basic guidance preprogrammed flight mode is "open loop," using gyro references, except as adjusted by BTL radio guidance commanded torquing of the gyros.

5.4.3.1 Flight Control Subsystem. Principal elements in the TAT autopilot system are the flight controller, programmer, rate gyros, and the hydraulic servos for actuating the engine gimbal actuators (Fig. 5-13). The flight controller consists of an attitude gyro package, ac amplifier-demodulator, shaping networks and a dc amplifier. The three HIG attitude gyros provide the prime inertial reference in the pitch, yaw, and roll planes, and are fitted with torquing coils for effecting programmed or guidance-commanded attitude changes. The ac amplifier-demodulator unit amplifies HIG gyro and rate gyro outputs and directs appropriate demodulated signals to the DC amplifier. The latter generates the proper signals for engine control hydraulic servo valve actuation, utilizing information provided by the ac amplifier and feedback signals from the valve actuators and rate gyros.

The control system programmer and timer unit is used to sequence preprogrammed flight events after liftoff. Thirteen channels are provided, of which nine are normally used for control functions.

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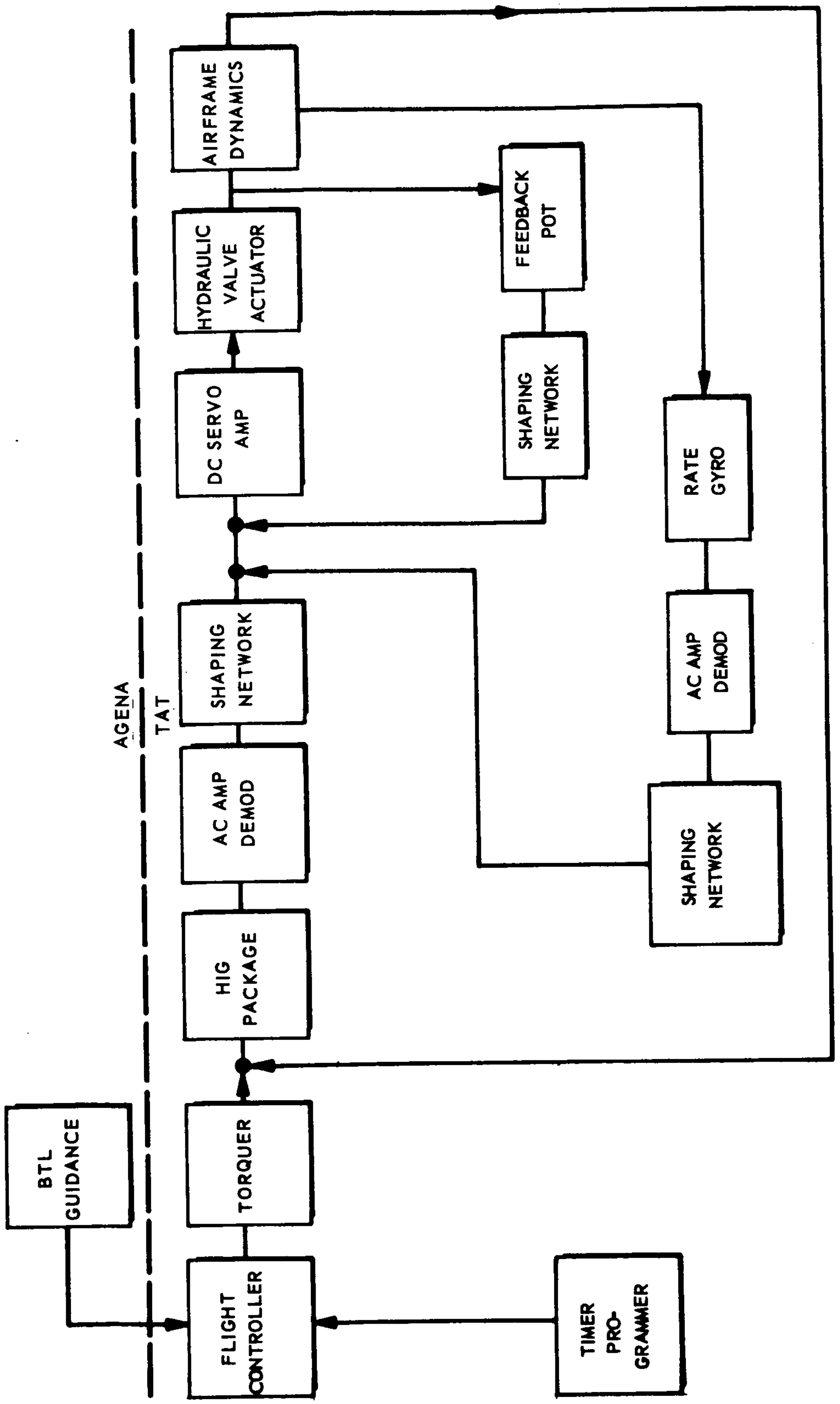

Figure 5-13 TAT Control System Block Diagram

5.4.3.2 BTL Guidance. The BTL Series 600 radio guidance system provided with the TAT Agena can be utilized for both the TAT and Agena phases (Agena first burn) of the boost ascent. The system consists of a ground guidance station with radar tracking and computer, and an airborne unit housed in the Agena and connected to the TAT by interstage wiring. The airborne guidance package consists of a receiver, decoder, and beacon transmitter. The unit is capable of receiving and decoding constant amplitude, time-modulated steering orders, as generated by the ground based computer, and which are used by the TAT flight controller to torque the attitude gyros. Cutoff discrete commands for terminations of thrust are also relayed by this command link. A pulse beacon signal is transmitted for use by the tracking radar which generates range, azimuth, and elevation information for the high speed computer. For some missions, this BTL guidance loop may also be utilized after TAT staging to control the Agena through first burn. (Limiting factor in Agena use is the decreasing tracker elevation angle as the vehicle proceeds downrange.)

5.4.4 Other TAT Airborne Systems

Other major TAT vehicle subsystems are:

a. Electrical Power — provides battery supplied 28v dc and 115/208v, 3-phase, 400 cps for inflight use, with provision for use of ground based supplies prior to liftoff.

b. Telemetry — provides instrumentation information on physical variables and component operation.

c. Hydraulic — includes the engine actuator systems.

d. Flight Termination — provides means of ground-commanded destruction of the vehicle in the event of unsafe flight condition.

SECTION 6
LOADS AND STRENGTH CRITERIA

6.1 GENERAL

This section summarizes the loads and strength criteria that are imposed on the spacecraft by the booster-Agena during ascent. These criteria are presented as an envelope of values from liftoff through injection into orbit for typical missions employing Atlas, Thor, and TAT boosters. The loads information applies specifically to the payload area and may be used by the spacecraft designer to define the load environment which would possibly be encountered with vehicles of similar configuration.

The loads presented consist of acceleration envelopes as a function of time for the total vehicle, load factors and elastic accelerations for the spacecraft adapter, and internal pressure histories for the payload area. This information has been compiled from flight histories, tests, and analytical predictions for the Mariner Mars, S-27, Comsat, Nimbus, EGO, POGO, Ranger, and OAO vehicles and is grouped according to the booster system employed for the launch. "General Environmental Specification for Equipment of the Agena and Associated Payload," LMSC-6117D, provides additional loads criteria which can be utilized as a guide by the spacecraft designer.*

6.2 DESIGN LOAD FACTORS

Flight accelerations which apply to the Agena/spacecraft adapter are presented in Tables 6-1, 6-2, and 6-3 for the three booster configurations — Atlas, Thor, and TAT. The tabulated "worst case" accelerations are composite upper limit

Note that LMSC-6117D is the approved issue of the Environmental Specification Document, and will be applicable, with exceptions, to all mission requirements.

or envelope values for previous Agena missions as represented by the vehicles mentioned in the table subheadings. They are defined at the Agena/spacecraft adapter, and therefore, represent the accelerations of the base of the spacecraft.

The first number quoted in the tables for any flight condition represents steady state (or rigid body) acceleration at the interface, and hence is the spacecraft steady state load factor. The second (±) number is the dynamic acceleration at the interface, and in order to find the dynamic load factor for the spacecraft, the spacecraft designer must consider the spacecraft and the spacecraft support structure. That is, if the rigidity of the spacecraft and spacecraft support structure is low enough to result in significant dynamic amplification over that of a rigid spacecraft, this amplification must be considered in the design loads. The rigidity required for minimum spacecraft amplification must be based on the overall vehicle dynamic characteristics. These dynamic characteristics are the low frequency modal responses described in Sections 7.4 and 7.5, and the Agena cutoff loads given in Section 7.7. The following guidelines may be used when defining preliminary loads for elastic spacecraft:

1. The most significant vehicle modes at max ($\alpha \bar{q}$), BECO (MECO), thrust tailoff, and post BECO (MECO) are usually at frequencies below 25 cps and 15 cps in longitudinal and lateral directions, respectively.
2. The MECO-8 oscillation may be considered a sustained sinusoid with a fixed frequency in the range of 19 ±3 cps.
3. The Agena engine cutoff shock spectra of Fig. 7-4 may be used to estimate longitudinal response with due consideration of the amplifications already included in the curves.

6.3 ACCELERATIONS

The total vehicle acceleration time histories for the three booster types being considered cover the time interval from liftoff through Agena final burnout. These acceleration histories shown in Figs. 6-1, 6-2, and 6-3, are presented as an envelope of nominal accelerations for each booster type (Atlas, Thor, TAT) and include, in cases of the Thor- and TAT-boosted configurations, the dynamic acceleration present at the Agena/spacecraft adapter during the 20 cps oscillation which starts approximately 15 seconds prior to booster engine cutoff. Figure 6-2 shows accelerations associated with a Thor/Agena for an Agena dual burn. Figure 6-3 shows accelerations associated with a TAT/Agena for an Agena single burn. The short dashed line provided above and below the acceleration curves represents the variation in acceleration created by tolerances on the propulsion system. These acceleration time histories are based on digital computer trajectory simulations modified with flight data where available.

The effect of flight conditions such as peak wind velocity on lateral acceleration, and of thrust "overshoot" (and ullage-rocket firing for Agena B ignition) on longitudinal and lateral acceleration have been incorporated in Tables 6-1, 6-2, and 6-3, and as such are not shown as part of the acceleration time history plots. The tables provide a listing of the critical flight conditions and the associated accelerations, both rigid body and dynamic, for the Agena/spacecraft adapter. The acceleration time histories are felt by the total vehicle with the exception of the 20 cps oscillation as mentioned above.

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Table 6-1

ACCELERATIONS FOR ATLAS/AGENA DEFINED AT SPACECRAFT/SPACECRAFT ADAPTER INTERFACE

(Mariner Mars, OAO, EGO, and Ranger Missions)

Event	Linear Acceleration		Angular Acceleration		
	Longitudinal (g)	Lateral (g)	Pitch (rad/sec^2)	Yaw (rad/sec^2)	Roll (rad/sec^2)
Max. $\alpha\bar{q}$	2.1 ± 0.3	0.3 ± 1.7	—	—	—
BECO	7.2 ± 0.5	0.05 ± 1.00	0.1	0.1	—
Thrust tail-off*	4.0	0 ± 1.0	—	—	0 ± 90
Post-BECO*	0 ± 0.2	0 ± 1.0	—	—	—
Agena 1st Ignition	1.3 ± 0.4	0.3 ± 1.0	1.6	—	—
Agena 1st Cutoff	2.4 ± 0.5	0 ± 1.0	—	—	—
Post-Agena Cutoff	0 ± 1.0	0 ± 1.0	—	—	—
Agena 2nd Ignition	2.7 ± 0.35	0.2 ± 1.0	1.75	—	—
Agena 2nd Cutoff	8.0 ± 0.5	0 ± 1.0	—	—	—
Ground Handling	Resultant of 2g in any direction				

Note: Values listed are a combination of "worst case" load factors for the combined vehicles listed above.

*In this context, Post-BECO is defined as zero thrust condition immediately following "thrust tail-off". Thrust tail-off refers to the time of the Atlas-Agena "torsional transit"

(±) = Oscillatory Acceleration

Table 6-2

ACCELERATIONS FOR THOR/AGENA DEFINED AT SPACECRAFT/SPACECRAFT ADAPTER INTERFACE

(Nimbus and Comsat Missions)

Event	Linear Acceleration		Angular Acceleration		
	Longitudinal (g)	Lateral (g)	Pitch (rad/sec^2)	Yaw (rad/sec^2)	Roll (rad/sec^2)
Max. $\alpha\bar{q}$	2.11 ± 0.30	0.4 ± 1.0	0.05	—	—
MECO – 8	6.54 ± 3.50	0.7 ± 1.0	0.106	—	—
Post-MECO*	0 ± 0.3	0 ± 1.0	—	—	—
Agena 1st Ignition	1.3 ± 0.4	0.11 ± 1.0	2.03	—	—
Agena 1st Cutoff	6.5 ± 0.5	0 ± 1.0	—	—	—
Post-Agena Cutoff	0 ± 1.0	0 ± 1.0	—	—	—
Agena 2nd Ignition	7.5 ± 0.4	0.63 ± 1.0	3.86	—	—
Agena 2nd Cutoff	7.5 ± 0.5	0 ± 1.0	—	—	—
Ground Handling	Resultant of 2g in any direction				

Note: Values listed are a combination of "worst case" load factors for the combined vehicles listed above.

*In this context, Post-MECO is defined as zero thrust condition immediately following thrust tail-off

(±) = Oscillatory Accelerations

Table 6-3

ACCELERATIONS FOR TAT/AGENA DEFINED AT SPACECRAFT/SPACECRAFT ADAPTER INTERFACE

(POGO Mission, Dual Burn)

Event	Linear Acceleration		Angular Acceleration		
	Longitudinal (g)	Lateral (g)	Pitch (rad/sec^2)	Yaw (rad/sec^2)	Roll (rad/sec^2)
Max. $\alpha\bar{q}$	1.7 ± 0.3	0.48 ± 1.00	—	—	—
MECO − 8	6.4 ± 3.5	0.06 ± 1.50	—	—	—
Post-MECO*	0 ± 0.25	0 ± 1.0	—	—	—
Agena 1st Ignition	1.3 ± 0.4	0.1 ± 1.0	2.0	—	—
Agena 1st Cutoff	5.8 ± 0.5	0 ± 1.0	—	—	—
Post-Agena Cutoff	0 ± 1.0	0 ± 1.0	—	—	—
Agena 2nd Ignition	7.1 ± 0.4	0.6 ± 1.0	3.5	—	—
Agena 2nd Cutoff	6.7 ± 0.5	0 ± 1.0	—	—	—
Ground Handling	Resultant of 2g in any direction				

Note: Values listed are a combination of "worst case" load factors for the combined vehicles listed above.

*In this context, Post-MECO is defined as zero thrust condition immediately following thrust tail-off

(±) = Oscillatory Accelerations

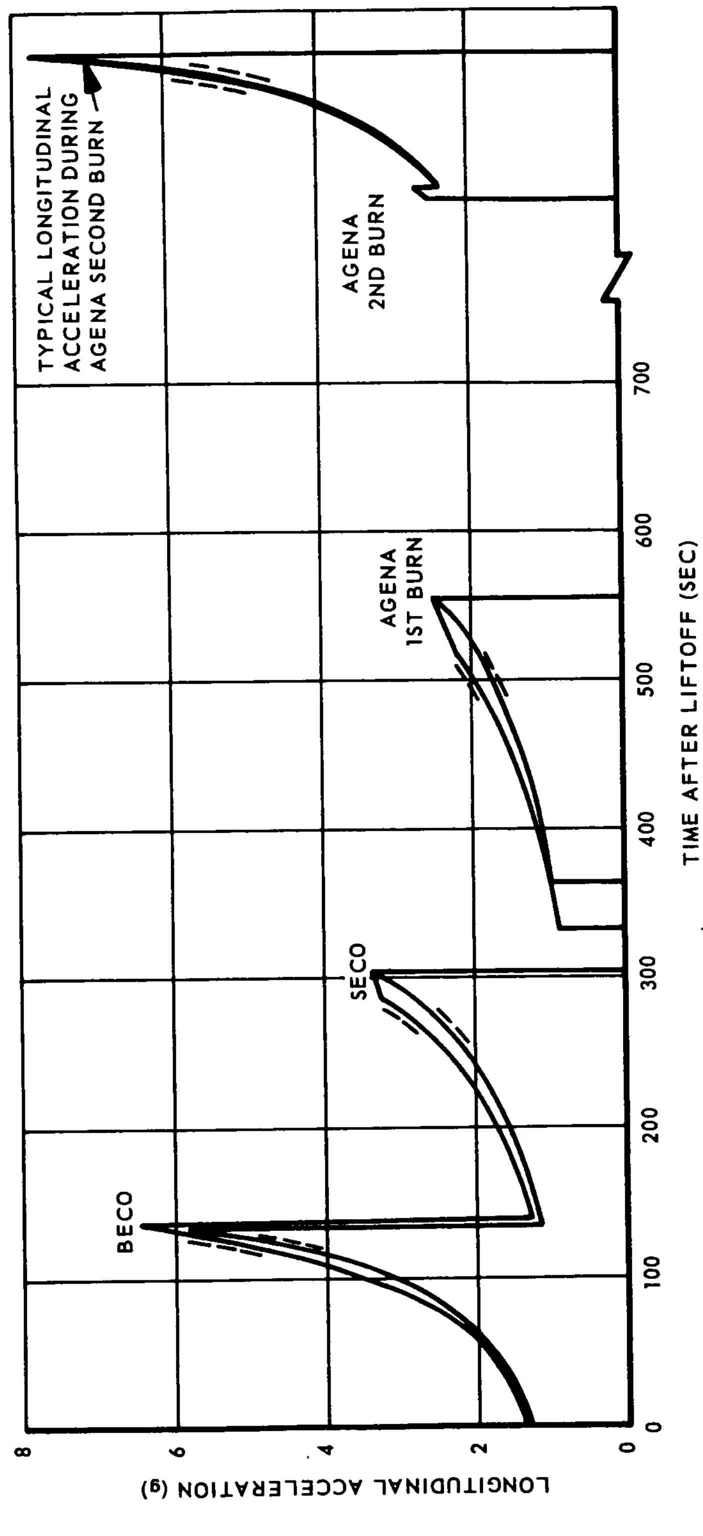

Fig. 6-1 Atlas/Agena Longitudinal Accelerations

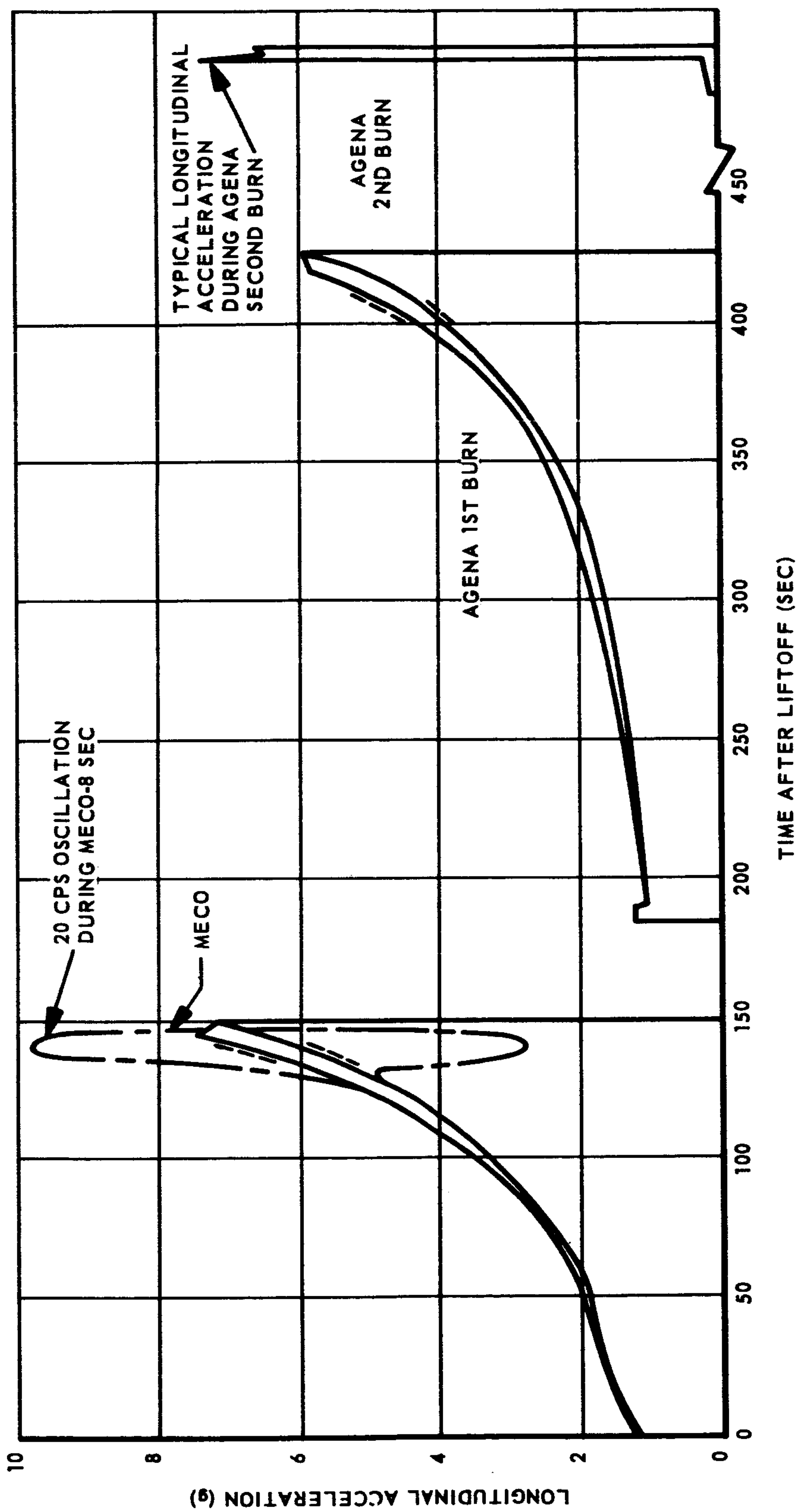

Fig. 6-2 Thor/Agena Longitudinal Accelerations

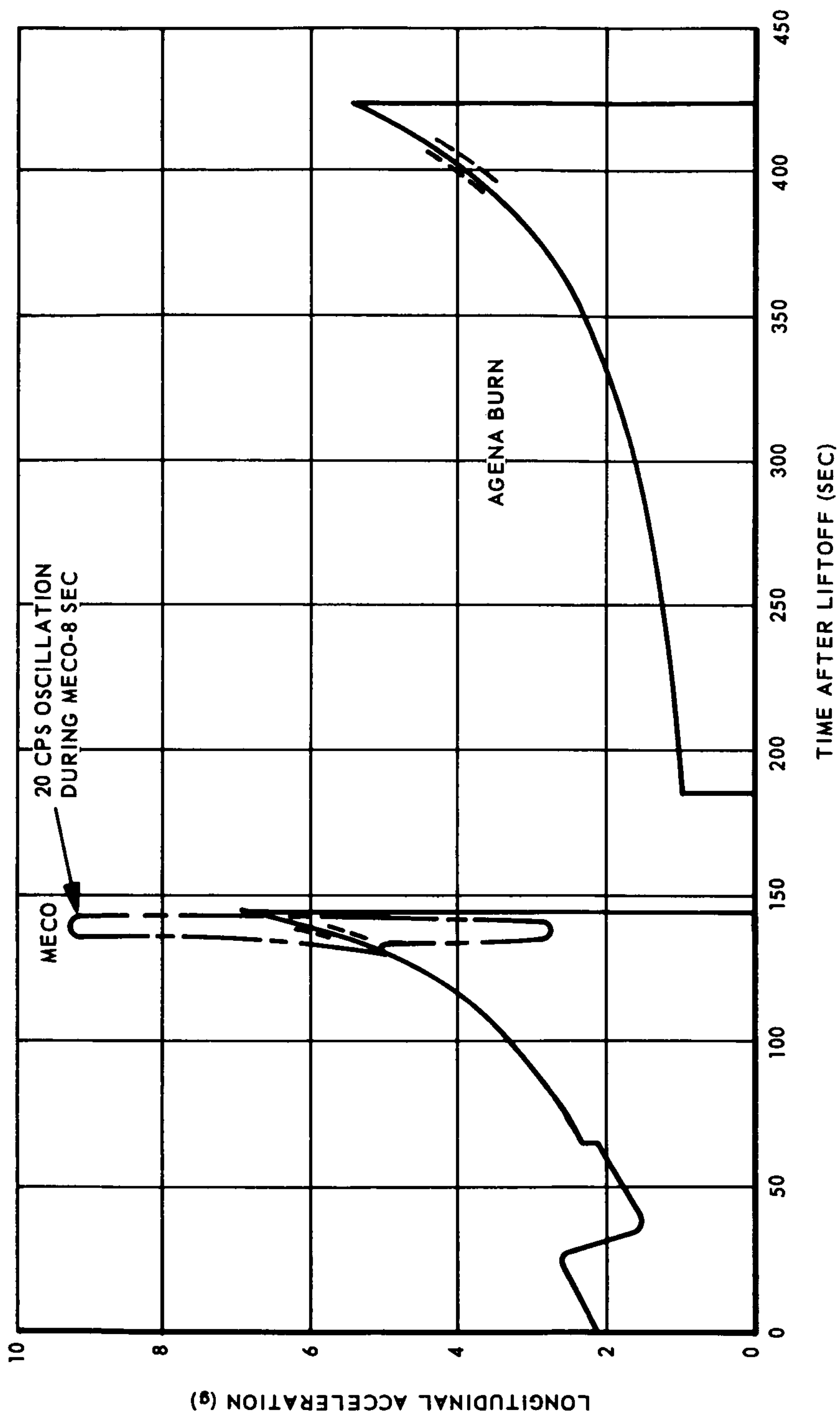

Fig. 6-3 TAT/Agena Longitudinal Accelerations

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6.4 SHROUD INTERNAL PRESSURE

The most significant factor affecting the shroud internal pressure of an ascent vehicle is the pressure history at the vent areas. This pressure history is, in turn, influenced by a number of factors such as location of an external vent area, dynamic pressure history, missile trajectory and winds. In addition, hole size, shroud volume, leakage areas, valve characteristics such as cracking pressures, acceleration or pressure locking and reseat pressures contribute significantly. Vent hole configuration is especially important because it dictates the value of the discharge coefficient. This discharge coefficient is also dependent upon flow direction, taking on a much different value for flow-out through an external vent into a cross stream as compared to flow-out through an internal vent into still air.

Agena internal rack pressure history also has a very significant effect on the shroud internal pressure history for virtually all vehicles. In many cases the only vent path for the shroud internal pressure is through the diaphragm into the forward rack through the fairings, and then out through the Agena longitudinal fairings to vent areas in the aft rack. On-pad internal pressure determination is governed essentially by air-conditioning parameters such as inlet pressures and flow rates. As this conditioned air is circulated and vented, virtually all the above factors are also significant.

Atlas- and TAT-boosted missions differ basically in trajectory. A TAT mission, because of its more rapid rate of ascent, yields a higher dynamic pressure history and consequently more extreme vent-hole pressure fluctuations. This yields a more extreme internal pressure history, resulting in higher burst or collapse pressures.

Venting of the Mariner Mars over-the-nose shroud (employing Atlas/Agena) is accomplished by one-way valves located in the diaphragm. Flow passes through the diaphragm into the forward rack, then through the two longitudinal fairings, and out through the vent areas provided in the aft rack.

A venting analysis was performed on the Mariner Mars vehicle and the resulting shroud internal pressure envelope is shown in Fig. 6-4. Ambient pressure history is given in Fig. 6-5. The steep pressure rates at approximately 11 seconds and 52 seconds reflect valve cracking and transonic effects, respectively. For analytical purposes the assumption is that, when cracking pressure is reached, the shroud cavity vent valve opens to its full area instantaneously.

The results of a venting analysis performed on the S-27, which employed a Thor/Agena, are given in Fig. 6-6. Ambient pressure history is given in Fig. 6-7. On this mission shroud venting is accomplished by open vent holes located in the shroud external skin as well as a one-way valve in the diaphragm which allows flow to pass from the shroud to the Agena forward rack only. A second one-way valve is installed in the diaphragm which allows flow from the Agena forward rack to the shroud cavity only. The second vent valve will open only when the pressure differential across the diaphragm (with excessive pressure in the Agena forward rack) approaches a level that will rupture the diaphragm. Here also, for analytical purposes, the assumption is that full area size occurs instantaneously when valve cracking pressure is reached.

6.5 SHROUD EXTERNAL PRESSURE COEFFICIENT DISTRIBUTIONS

The aerodynamic difference of the various shrouds is presented in the form of pressure coefficient distributions. These distributions for Nimbus, Standard Agena Clamshell Shroud, Ranger, and Mariner Mars are shown in Figs. 6-8 through 6-13. The vehicles for these programs represent the three major shroud configurations discussed in Section 10. The OAO Shroud has generally the same configuration as the Nimbus type shroud except for the base diameter. Therefore, the pressure coefficient distributions shown for the Nimbus Shroud cone cylinder combination will be similar to those of the OAO.*

Distributions for each shroud are shown for Mach numbers in the range of 1.0 and 1.4 at a nominal angle of attack of 6.0 degrees. Each figure shows the angle of attack, Mach number, and shroud shape to which the pressure coefficient distribution applies. These distributions were taken from reports of wind-tunnel tests which are referenced on the applicable figures.

*Wind tunnel data is not available for the OAO Shroud.

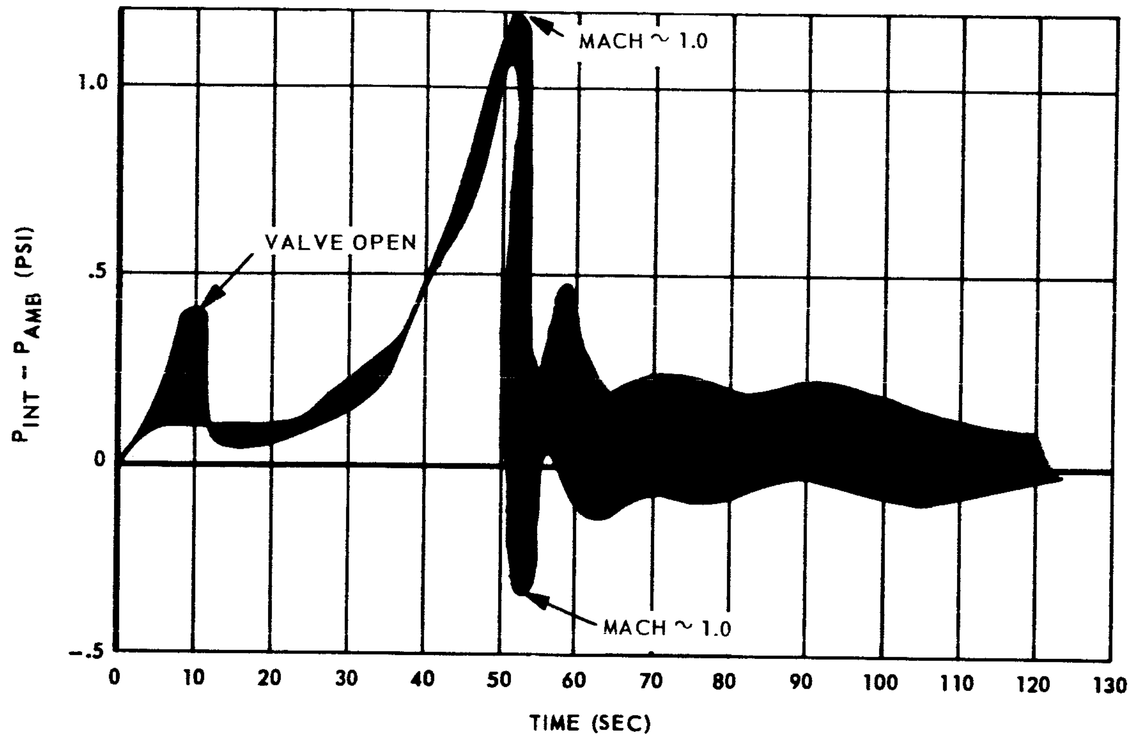

Fig. 6-4 Shroud Internal Pressure History Envelope, Model 43205 (Mariner Mars)

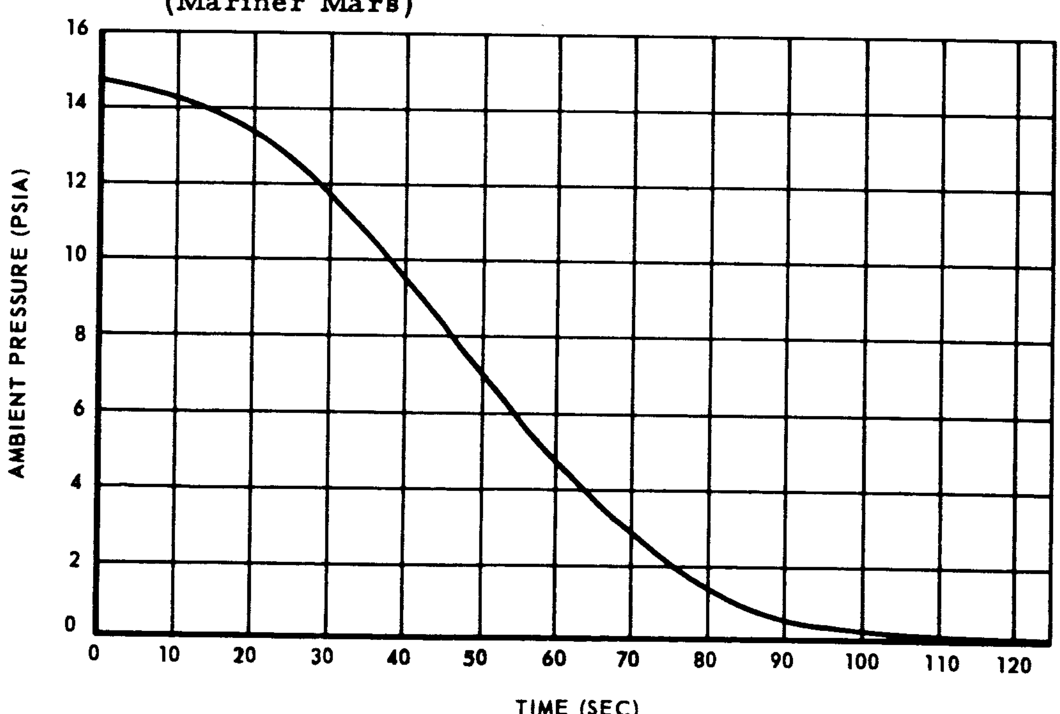

Fig. 6-5 Shroud Ambient Pressure History, Model 43205 (Mariner Mars)

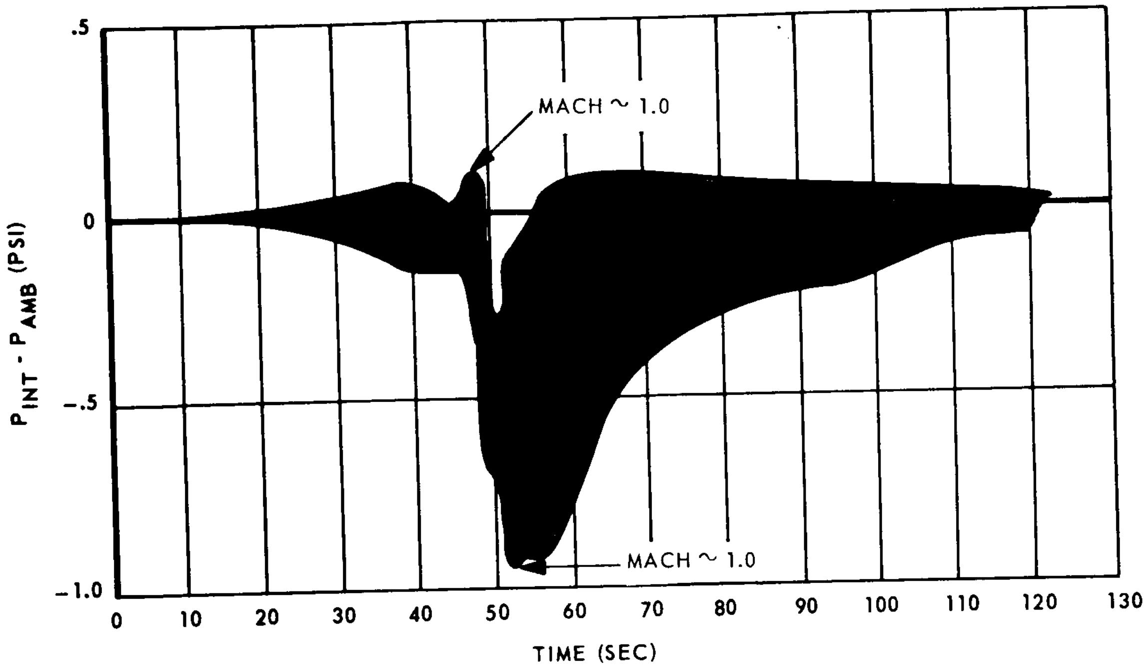

Figure 6-6 Shroud Internal Pressure History Envelope, Nimbus Type

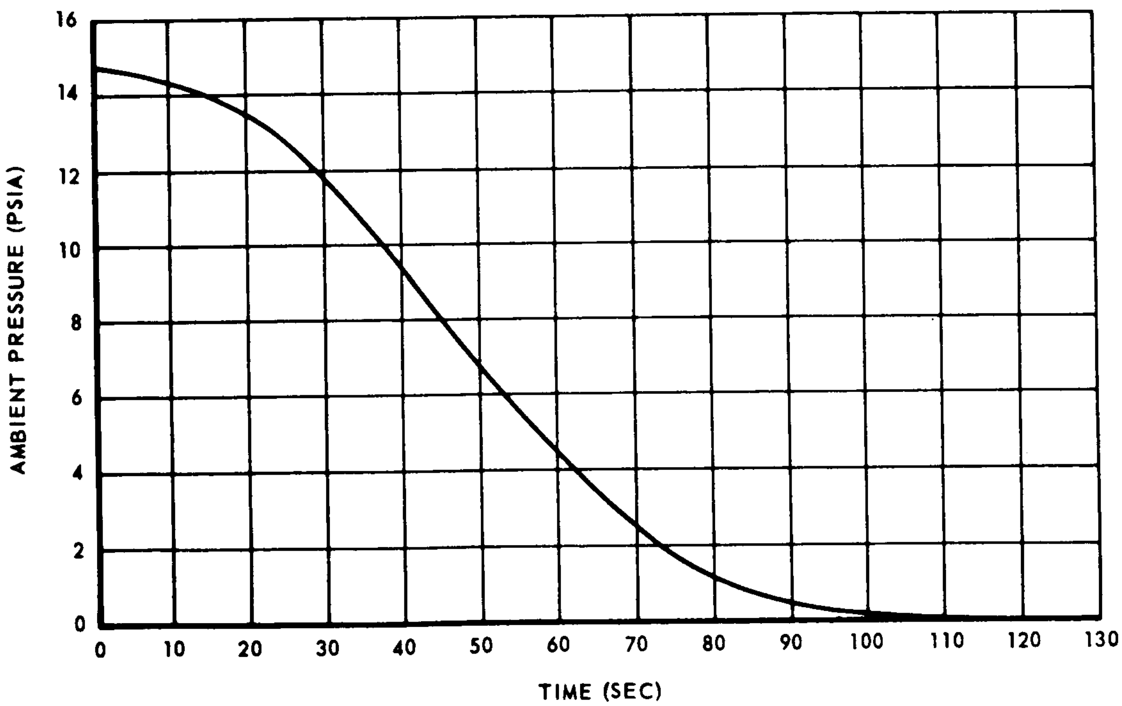

Figure 6-7 Shroud Ambient Pressure History, Nimbus Type

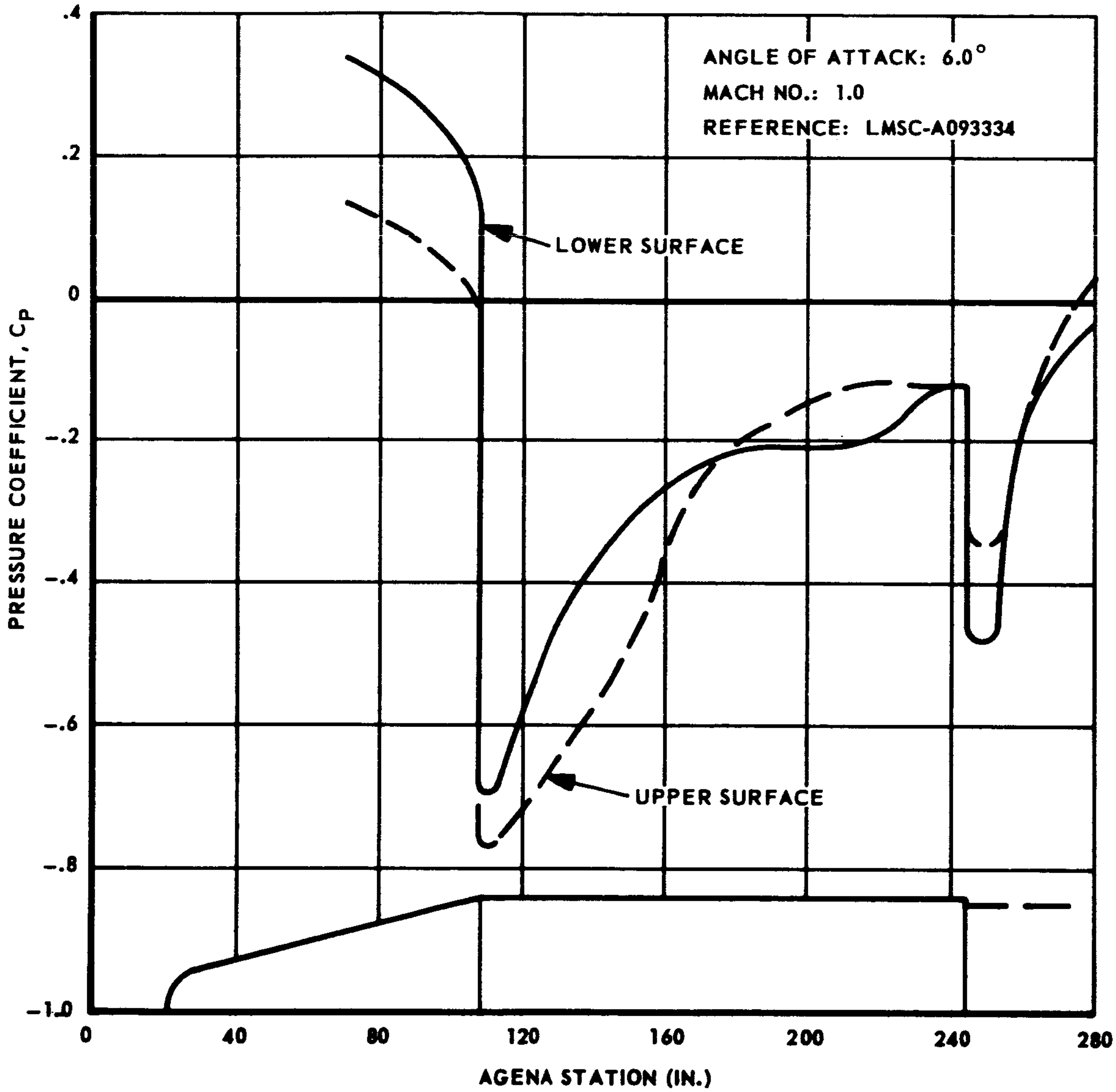

Fig. 6-8 Nimbus and Standard Agena Clamshell Shroud, Mach 1.0 Pressure Coefficient Distribution

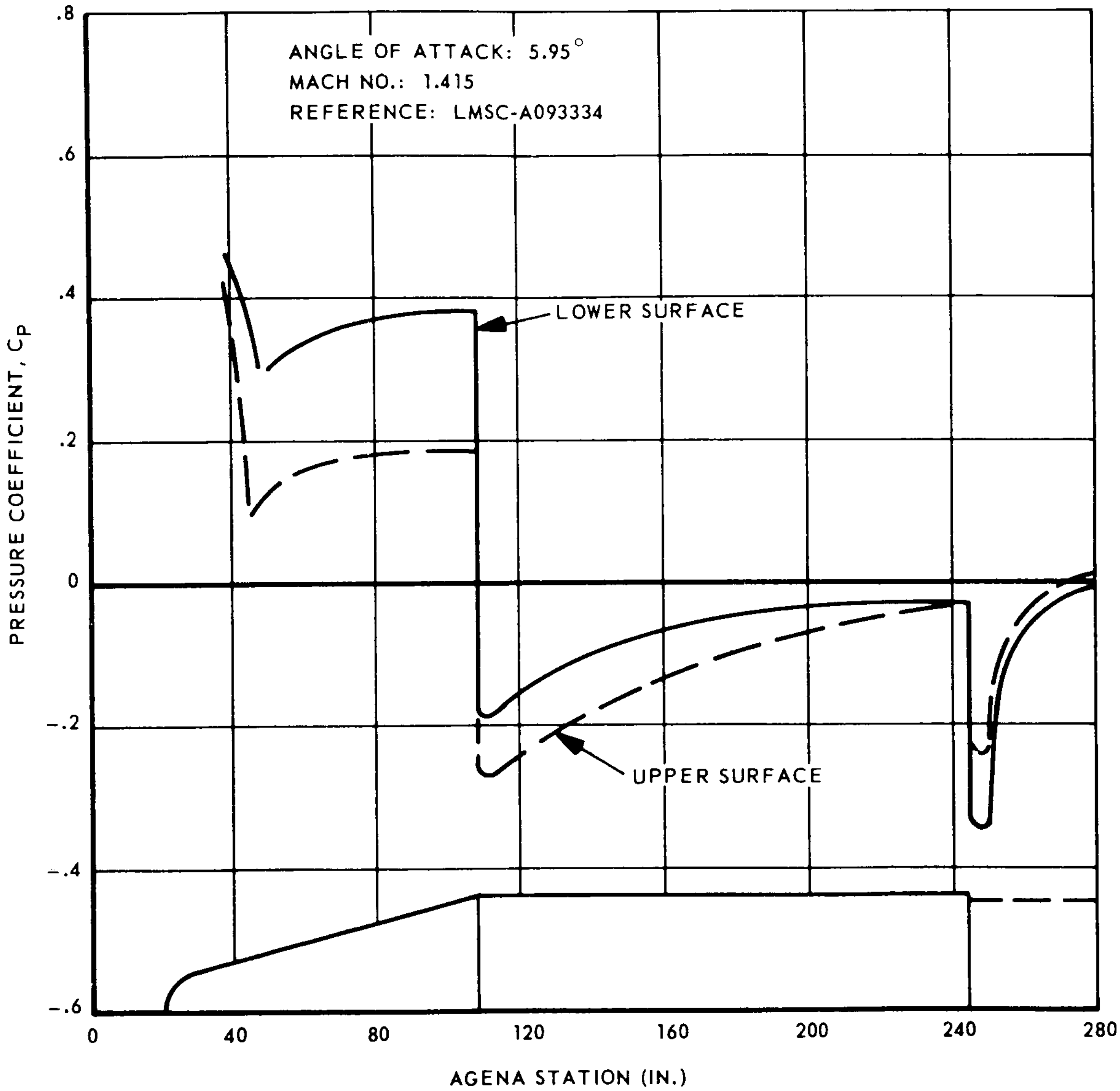

Fig. 6-9 Nimbus and Standard Agena Clamshell Shroud, Mach 1.4 Pressure Coefficient Distribution

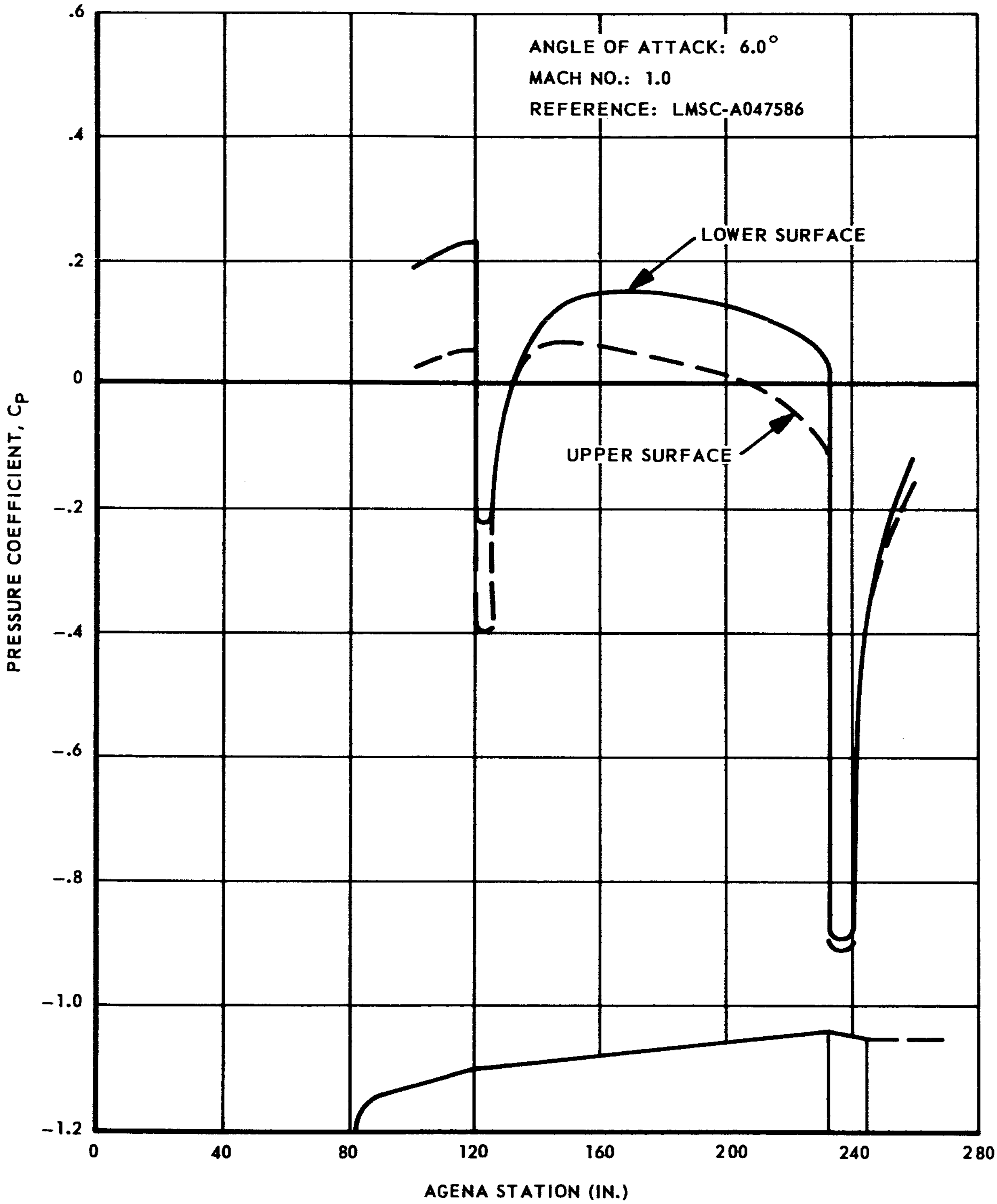

Fig. 6-10 Ranger Shroud Mach 1.0 Pressure Coefficient Distribution

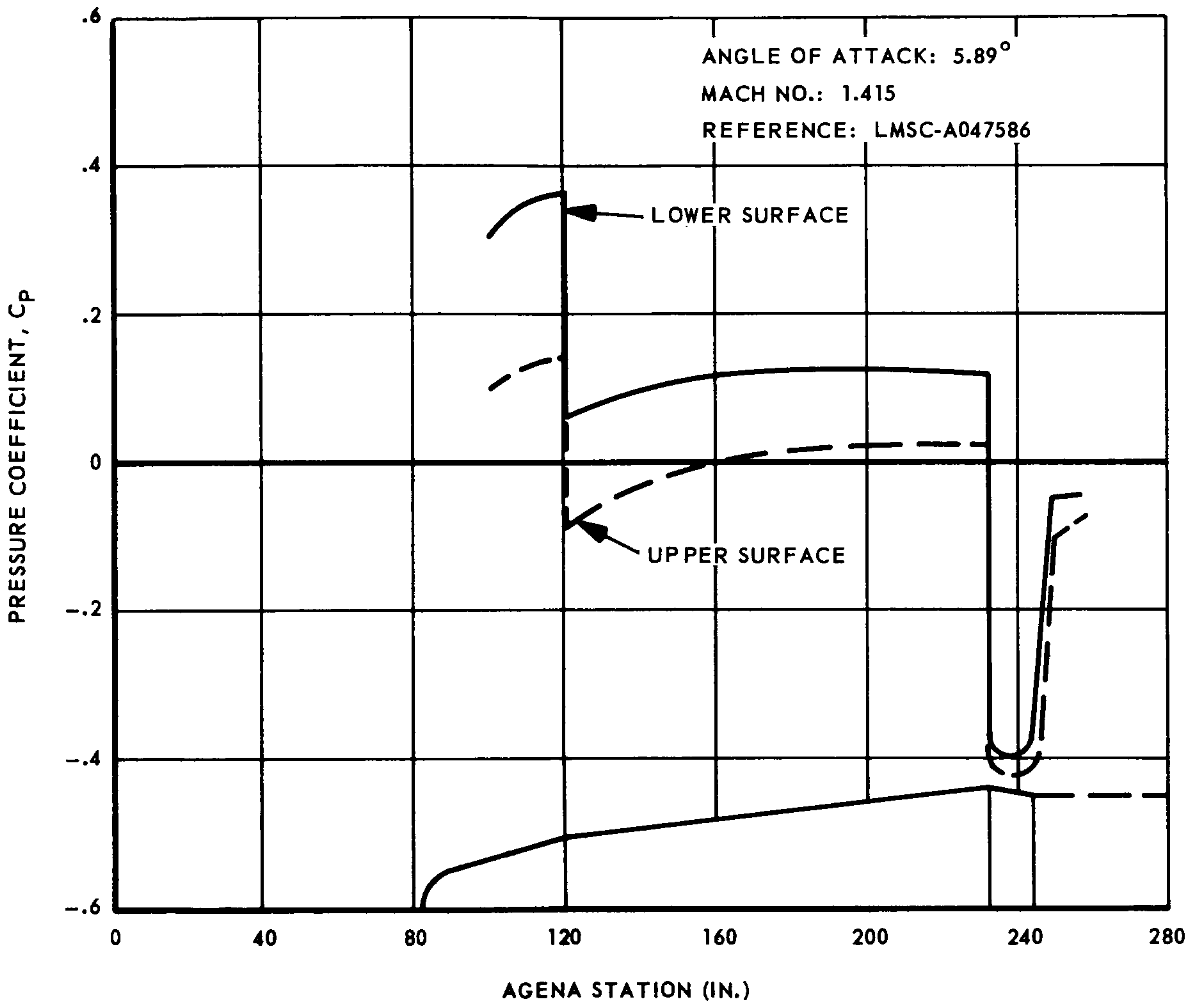

Fig. 6-11 Ranger Shroud Mach 1.4 Pressure Coefficient Distribution

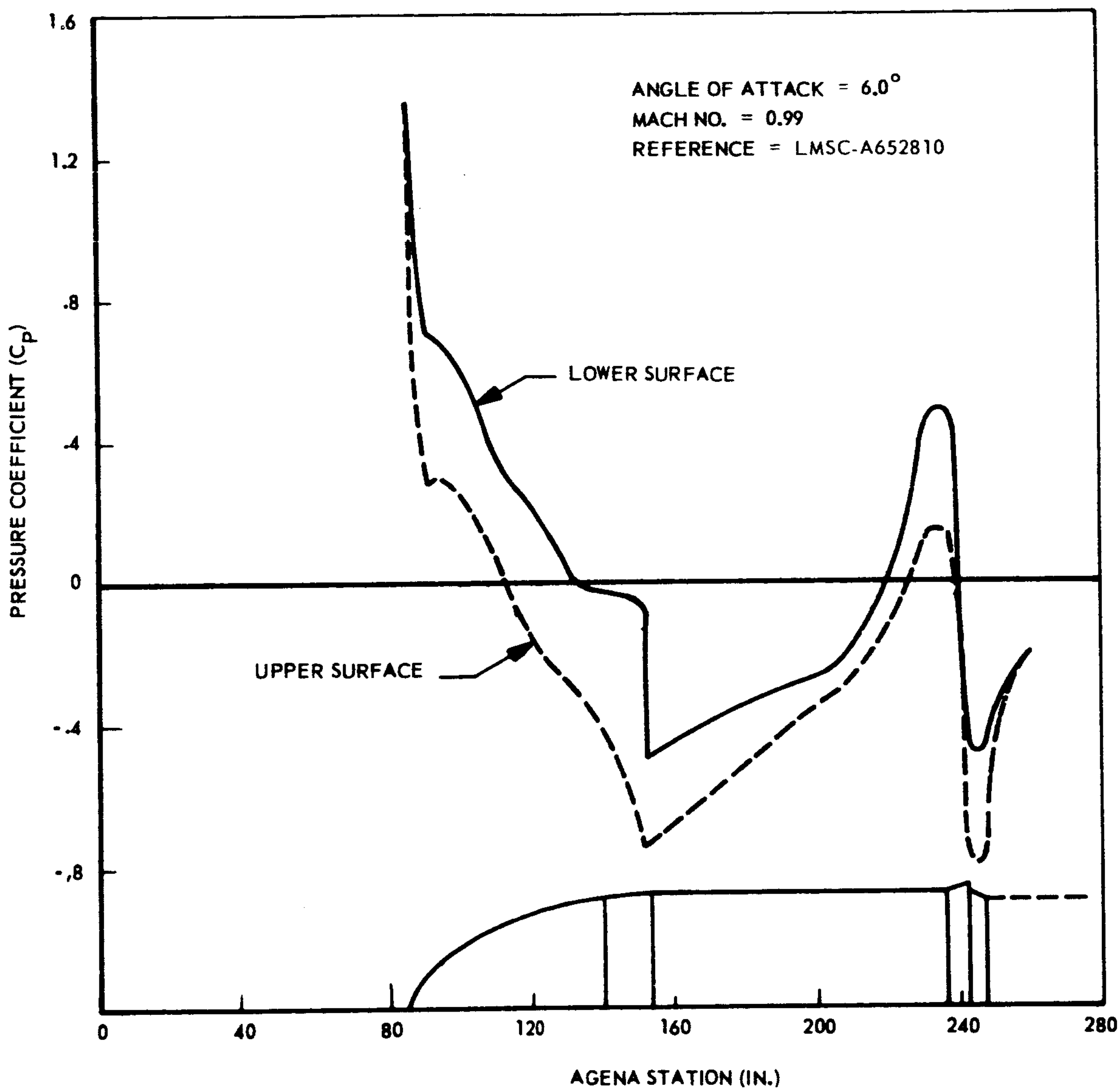

Fig. 6-12 Mariner Mars Shroud Mach 0.99 Pressure Coefficient Distribution

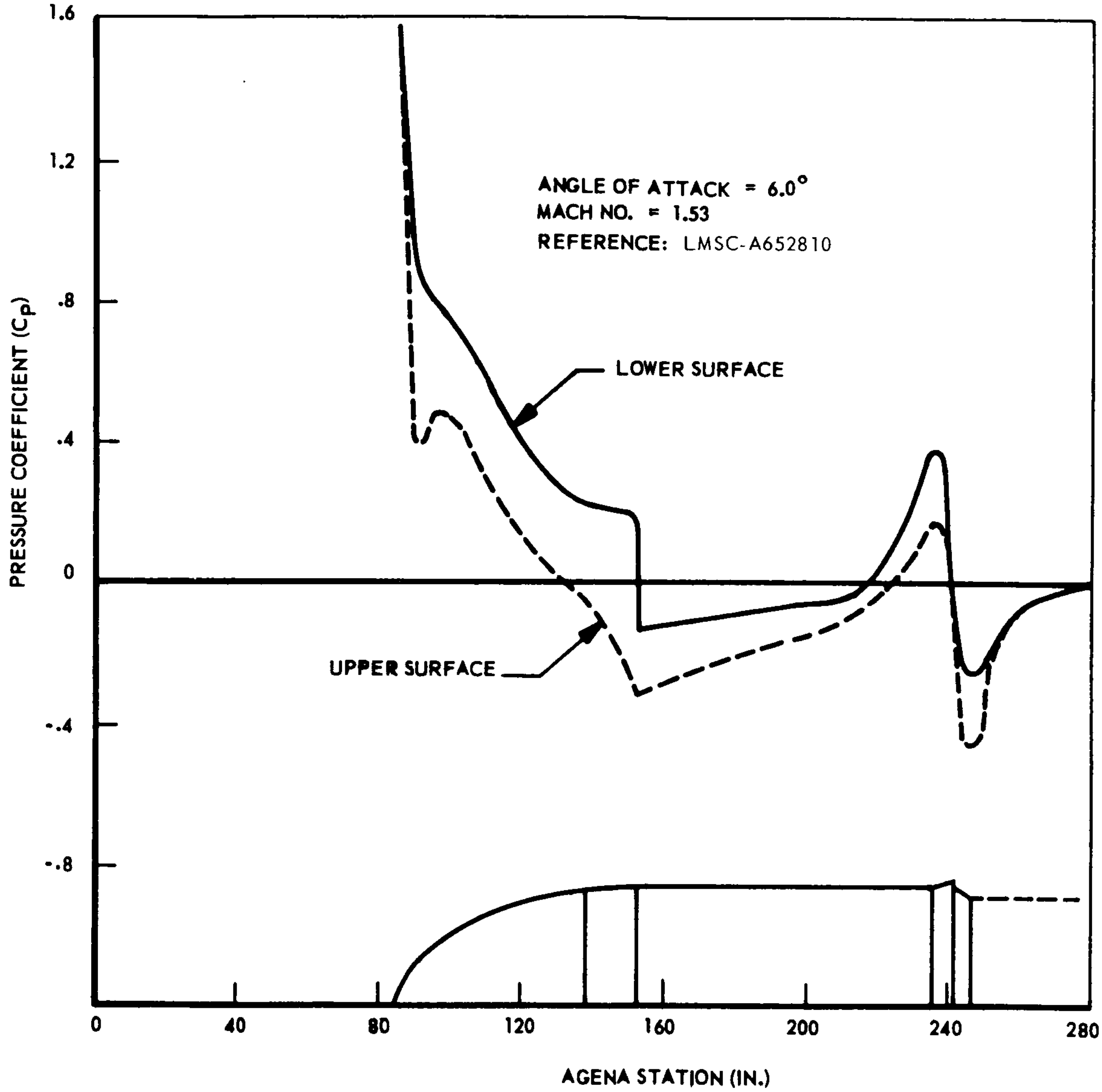

Fig. 6-13 Mariner Mars Shroud Mach 1.53 Pressure Coefficient Distribution

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Section 7
DYNAMIC CRITERIA

7.1 GENERAL

This section summarizes the dynamic environmental criteria that are imposed on the spacecraft by the Agena/booster vehicle during launch and ascent. Dynamic environments are presented as an envelope of values from liftoff through injection into orbit for typical missions employing Atlas, Thor, and TAT boosters. This data was compiled from flight histories, tests, and analytical predictions. LMSC-6117D, "General Environmental Specification for Equipment of the Agena and Associated Payload," provides additional dynamic criteria which can be utilized as a guide by the spacecraft designer. *

The dynamic environment of the Agena/spacecraft is one of combined mechanical, aerodynamic, and acoustic excitation. The excitation sources are varied with the more prominent ones being propulsion system perturbations, exhaust stream noise, boundary layer turbulence, oscillating shock waves, and staging transients. A number of measurements of this environment have been made and a summary of this data is given in the following paragraphs.

7.2 ACOUSTIC ENVIRONMENT

Time-history plots of the internal and external acoustic environment during the launch and ascent phase of flight are given in Fig. 7-1. The measurements were obtained from a TAT-boosted spacecraft and represent the most severe acoustic environment observed to date. Table 7-1 summarizes the external sound-pressure levels generated at liftoff by TAT, Thor, and Atlas boosters. Internal sound-pressure levels can be conservatively assumed to be 10 db below the external levels. The limited number of internal acoustic measurements made precludes an accurate description of this environment.

*Note that LMSC-6117D is the approved issue of the Environmental Specification Document, and will be applicable, with exceptions, to all mission requirements.

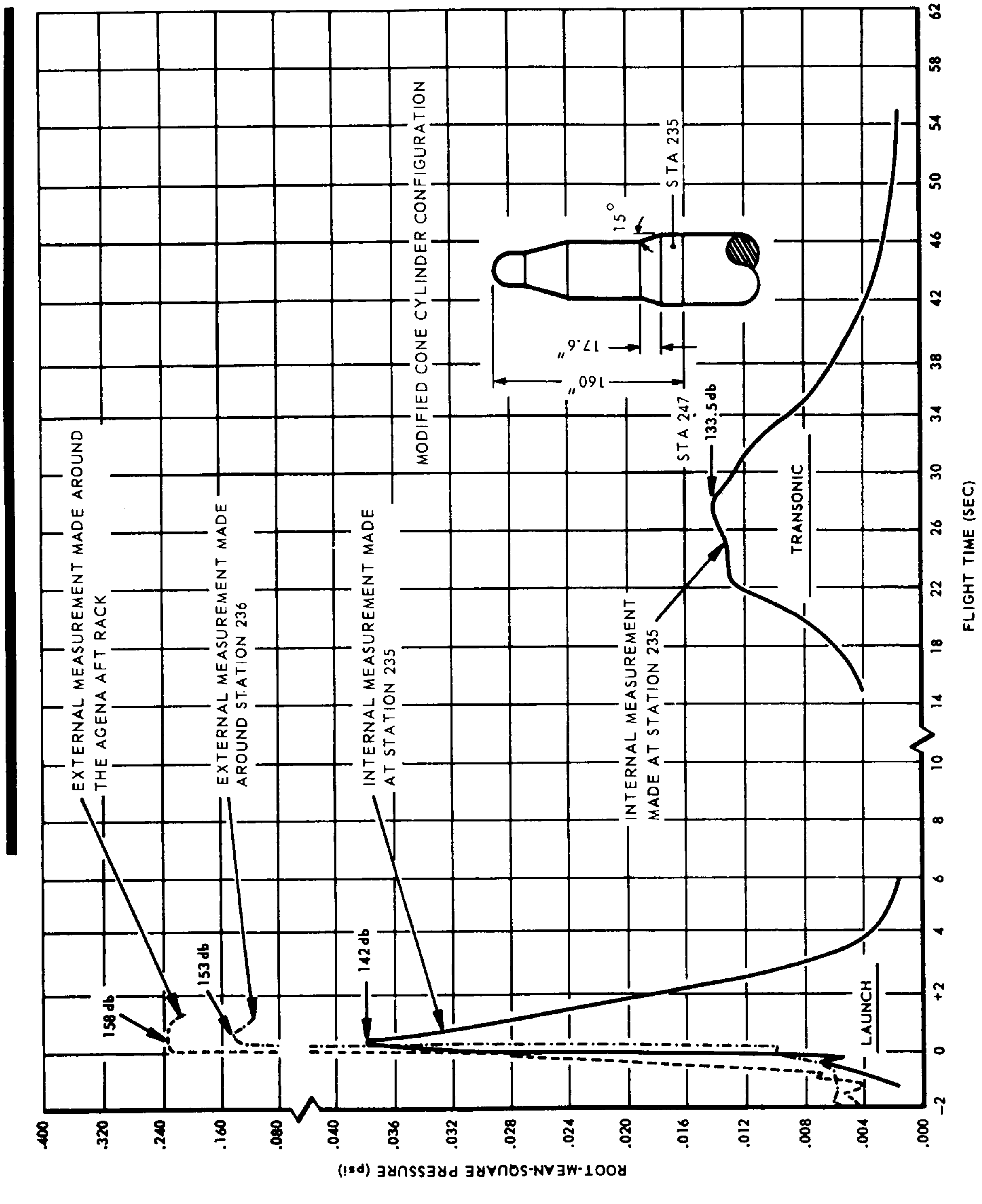


Fig. 7-1 TAT-Boosted Spacecraft Acoustic Pressure Time History

Table 7-1

SUMMARY OF MEASURED EXTERNAL SOUND-PRESSURE LEVELS AT LIFTOFF (TAT, Thor, Atlas)

Location	Station	TAT		Thor	Atlas
		Dry Pad	Wet Pad	Dry Pad	Wet Pad
Payload and Auxiliary Equipment Racks	138.5				147
	153.5				147.5
	200	153	148		
	233			150.5	
Forward Equipment	267*	155	150	151	147.5
	300	156	151		
Aft Equipment Racks	435*	157	152	152	147.5
	460	158	152		
	462.5				147.5
	474.5				148
Booster Midsection	617			154	
	761			154	
	770	161	154		

Sound-pressure levels are in db. (reference level: 0.0002 Dyne/cm^2)

*Stations for which values were found by interpolating from measured values.

7.3 HIGH FREQUENCY ENVIRONMENT

The high frequency spacecraft vibrations induced by the random acoustic pressure fields are illustrated in Fig. 7-2. The plots presented in this figure can be used only to show trends since the measurements represented, which are typical of those that are available, were obtained from different locations on the vehicle structure. Because a vibration pickup will measure any structural response, the measurement will be strongly dependent on the instrument's structural installation. However, the following trends are clearly evident:

a. When a spacecraft is launched from a "dry" pad, the acoustically induced vibrations are more severe than when the launch is from a "wet" pad*. This is attributed to the increase in noise reflected upwards from the exhaust deflector plates; refer to Table 7-1.

b. When a hammerhead shroud configuration is used, the transonic structural response is more severe than that experienced during the first few seconds of an engine ignition.

c. The transition angle between the hammer-head shroud and Agena diameter has a considerable influence on the duration of the transonic excitation.

A direct comparison of the high-frequency vibration levels induced in TAT-, Thor-, and Atlas-boosted spacecraft is extremely difficult with the data available. However, since it is primarily the local external noise that generates this environment, the most severe vibration levels will be induced in a TAT-boosted spacecraft flying a hammerhead nose shroud and comparatively lower levels will be induced in Atlas-boosted spacecraft flying conventional cone-cylinder nose shrouds.

*A "wet" pad has a water-cooled exhaust bucket with the exhaust from the propulsion system being directed along an irrigated trench. A "dry" pad utilizes only exhaust-deflector plates which are not water-cooled.

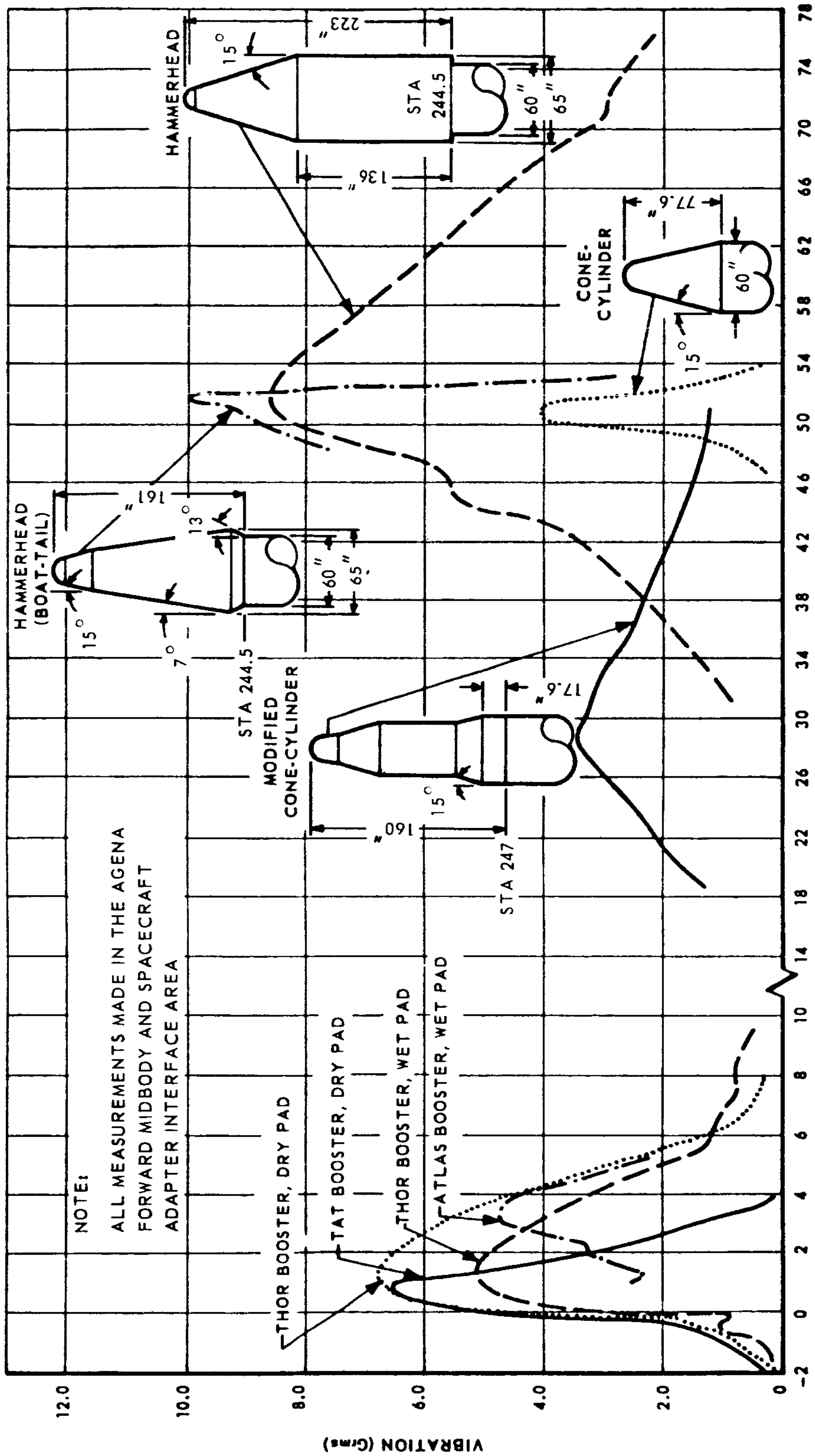


Fig. 7-2 Typical Spacecraft Vibration Time History

7.4 LOW-FREQUENCY ENVIRONMENT

The flight events which cause the most significant low-frequency structural excitation are launch, booster engine cutoff, and Agena engine cutoff. The last two events are discussed in pars. 7.5 and 7.7 and only the launch event is discussed in this paragraph.

On Atlas-boosted flights the second and third structural bending modes of the vehicle are generally excited. The frequency of these modes are about 6 and 11 cps respectively, and have been measured in the forward rack area with an amplitude of 0.5 g 0-to-peak. A transient type of longitudinal oscillation, involving coupling between the Atlas propellant tank, ullage pressure, and regulator system, will also be present. The frequency of this oscillation is about 5 cps and has an amplitude of approximately 0.5 g 0-to-peak.

On TAT- or Thor-boosted flights, the first longitudinal and first lateral bending modes of the vehicle are most prominent. Responses in these modes have been measured in the forward rack area with amplitude of 1 and 0.25 g 0-to-peak respectively. The frequency of the longitudinal oscillation is approximately 16 cps, and that of the lateral oscillation is about 3 cps.

7.5 BOOSTER RESONANCE PHENOMENON

A longitudinal oscillation will occur on TAT/Agena or Thor/Agena flights near the time of booster burnout. Although the exact cause is not completely understood, this oscillation is attributable to some form of engine-structural coupling. The main characteristics of this oscillation are that it is predominately in the longitudinal direction; it initially appears during the last 30 to 40 seconds of the boosted flight phase, it dies down before booster burnout; it exhibits a frequency of approximately 20 cps which is close to the first longitudinal mode of the vehicle at this time of flight; and the magnitude of this oscillation has been measured in the aft rack area as 2.3 g 0-to-peak. Based on the modal characteristics of this vehicle, it was determined that this disturbance would reach 3.2 g 0-to-peak in the forward rack.

During the main engine cutoff of the Atlas booster a high amplitude 70 cps torsional transient is induced in the Agena structure. Again, the source of this oscillation has not been clearly defined, but there is some evidence that combustion instability in the engine during thrust decay is responsible. This transient has been measured in the forward rack of the Agena with a peak-to-peak amplitude of 9.6 g, and on the booster adapter with a peak-to-peak amplitude of 5.7 g. These measurements were obtained from tangentially oriented accelerometers; located at a distance of approximately 30 inches from the longitudinal axis of the spacecraft. Figure 7-3 presents shock spectra, i.e., the maximum response of simple single-degree-of-freedom systems to a transient excitation, for this transient.

7.6 STAGING TRANSIENTS

Pyrotechnic devices are used to initiate events such as booster separation, shroud separation, and jettisoning of the Agena horizon sensor doors. When these devices are fired, a very high amplitude shock pulse of 5 to 10 milliseconds duration is generated and travels through the structure. Table 7-2 lists the measurements that have been made of this transient.

The effect of pyrotechnic shock has only recently been considered, and is still under investigation. Preliminary conclusions are that the pyrotechnically induced shocks have no appreciable effect on primary vehicle structure, secondary structure, or equipment structural hardware. The effect on electronic, optical, or precision mechanical components of equipment and subsystems has not been established; however, no failures have been attributed to this cause.

7.7 AGENA-POWERED FLIGHT ENVIRONMENT

During the Agena-powered flight phase the dynamic environment is less severe than that of the launch and booster ascent phase. This is due to the low-density atmosphere encountered at altitudes of Agena-powered flight.

The most significant structural excitation is due to the transients that are generated at Agena engine cutoff. Shock spectra of the transients measured in the forward and aft equipment racks are given in Figs. 7-4 and 7-5.

7.8 GROUND TEST FIRING ENVIRONMENT

Broad-band random acoustic pressure fields, generated by the exhaust stream of the Agena propulsion system,induce high frequency (structural vibrations) during ground test firings. Levels as high as 26 g rms have been measured on the aft rack structure, and up to 6.8 g rms on the forward rack structure. The associated acoustic levels were 156 and 145 db around the aft and forward racks respectively. Superimposed on the broad-band excitation in the aft rack area are very high, narrow-band excitations that reach levels of 22.7 g rms. They are consistently found centered around 375, 450, 920, 1650, and 1880 cps, and are believed to be associated with the propellant pumping system and engine combustion resonances.

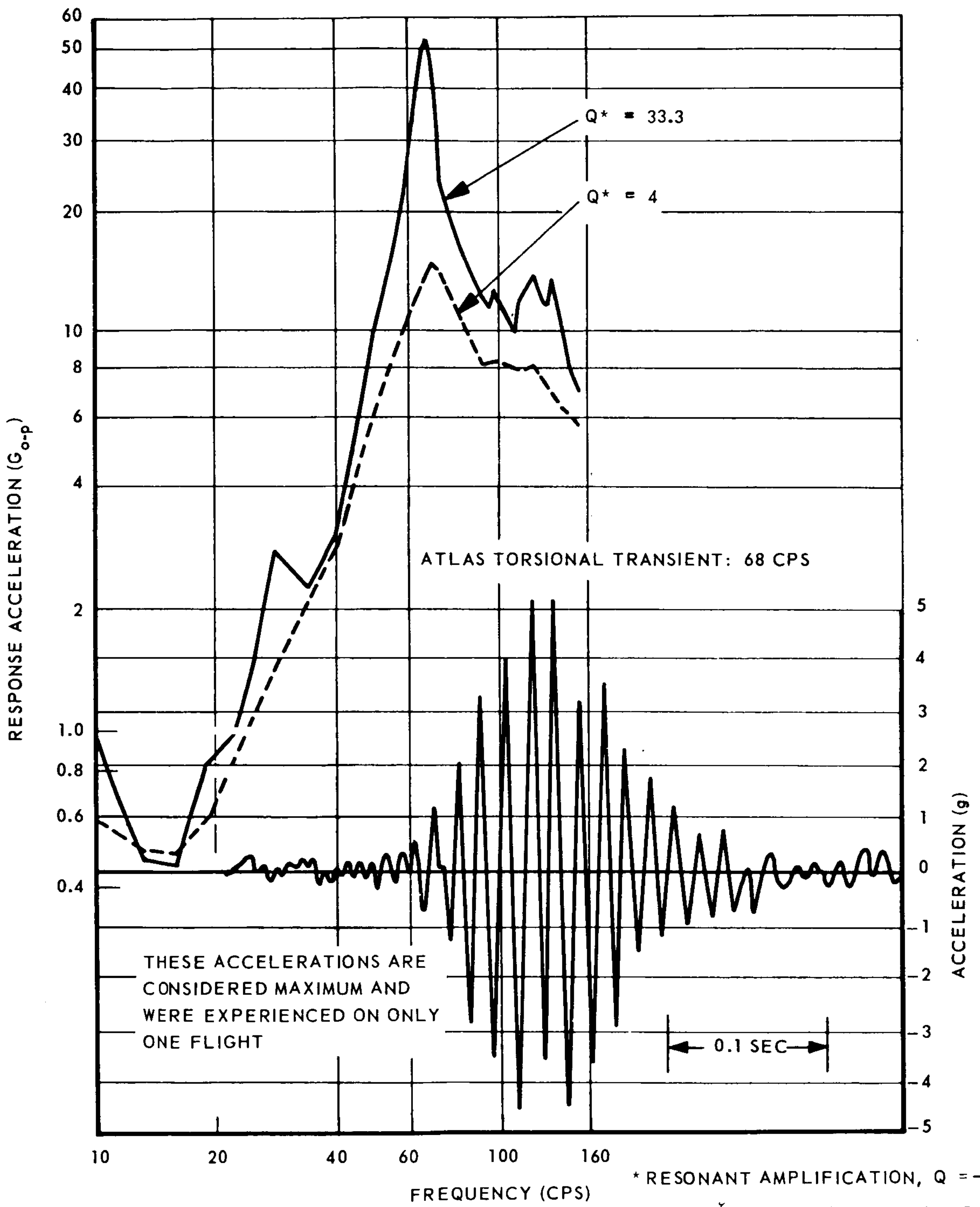
7.9 QUALIFICATION TESTS

The measurements that have been made of the flight and ground test environments were used to establish test levels and design philosophy for the Agena structure. Measurements from which the test levels were obtained were primarily accelerometers (strain gage and piezoelectric types) installed on structural "hard-points" in the Agena forward and aft equipment racks, and in spacecraft adapters. Envelopes of vibration levels were derived and taken to represent the "limit" structural environment. These measurements in combination with engineering judgment and past successful experience were also used to establish equipment dynamic qualification test levels. The equipment levels are set higher than the structural levels to account for mounting bracket resonances.

The resulting qualification test levels for which equipment and structural bracketry must be designed are summarized for typical envelopes in Tables 7-3 through 7-8. The equipment test levels are for equipment weighing 75 lb or less and represent an envelope of the maximum levels specified for different locations within the Agena. Equipment test levels are described in greater detail in LMSC-6117D.

The structural test levels envelope the flight measurements and represent an upper bound for this environment. Therefore, relief in individual frequency bands can be given at the resonance of major equipment and structural modes.

For preliminary design purposes, the spacecraft designer should assume that the forward rack structure and equipment vibration envelopes given in Tables 7-3, 7-4, and 7-5are the minimum levels the spacecraft would expect to see. He should then analyze the spacecraft as indicated in paragraph 6.2 to obtain dynamic loads. Because of well known narrow-band flight disturbances, it is recommended that the structure and equipment of the Agena be designed to avoid coincident resonance frequency conditions in the vicinity of 20, 70, 375, 450, 920, 1650, and 1880 cps.

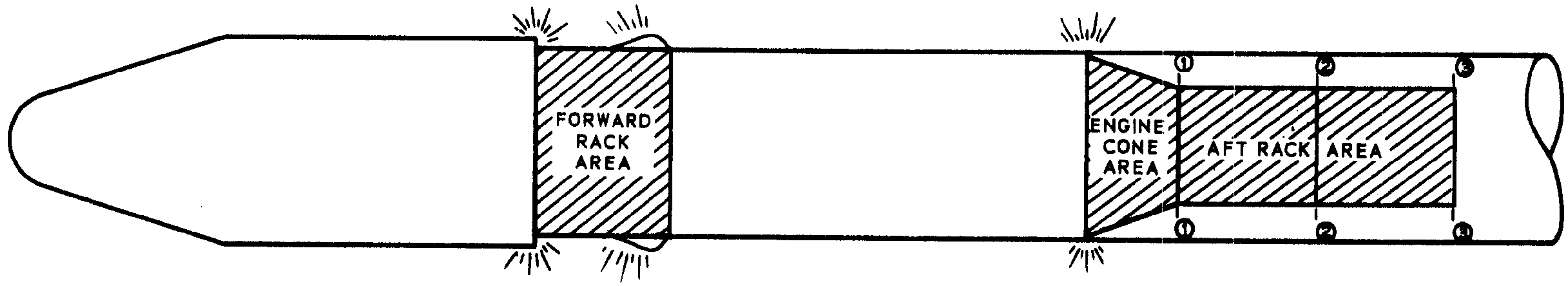


* RESONANT AMPLIFICATION, $Q = \frac{1}{2(\zeta)}$
WHERE ζ IS THE DAMPING FACTOR

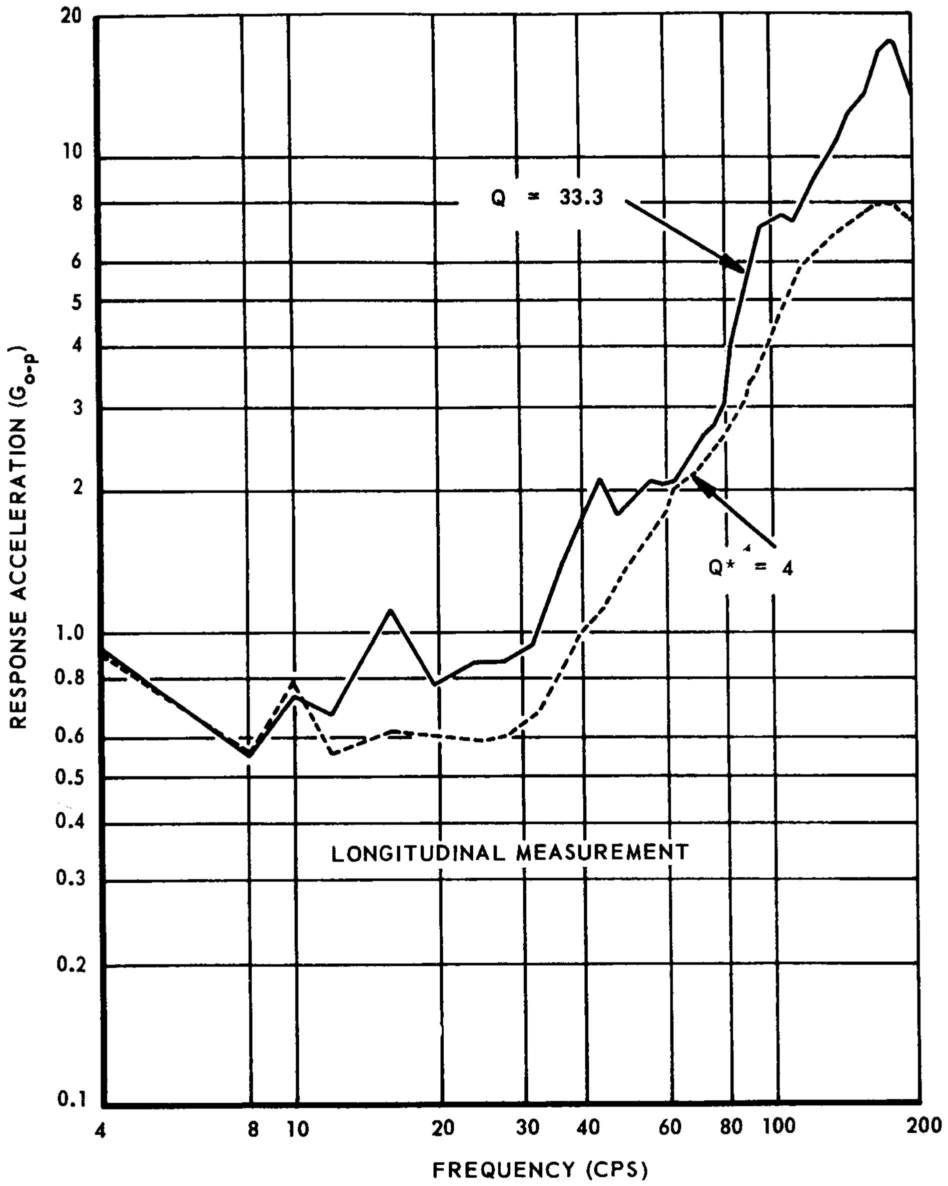
Fig. 7-3 Shock Spectra of Atlas Torsional Transient, Forward Equipment Rack, Station 232

Table 7-2

MEASURED PYROTECHNIC SHOCK



Event	Fwd. Section		Aft Section							
	Fwd. Rack		Engine Cone		Eng. Mounting Ring ①–①		Aft Rack ②–②		Aft Rack ③–③	
	G's (0-p)	Freq.	G's (0-p)	Freq.	G's (0-p)	Freq.	G's (0-p)	Freq.	G's (0-p)	Freq.
Booster Separation	40 200 140 – – 300 – 300	900 1300 2000 – – 5500 – 6500	3000 – – – 2000 – – –	900 – – – 5000 – – –	800 – 1100 – 600 – 1500 –	900 – 2000 – 5000 – 6000 –	– – – 400 – – – –	– – – 4000 – – – –	130 – 140 – – – – –	900 – 2000 – – – – –
Shroud Separation	280 950 2000 2800 400 1500	400 1000 2000 2500 5600 7500	← No Measurements →							
Horizon Sensor Door	230 150 200 400	500 1000 4000 6500	← No Appreciable Levels Measured Due to this Event →							



$$* \text{RESONANT AMPLIFICATION, } Q = \frac{1}{2(\zeta)};$$

WHERE ζ IS THE DAMPING FACTOR

Fig. 7-4 Shock Spectra of Agena Engine Cutoff, Forward Equipment Rack, Station 245

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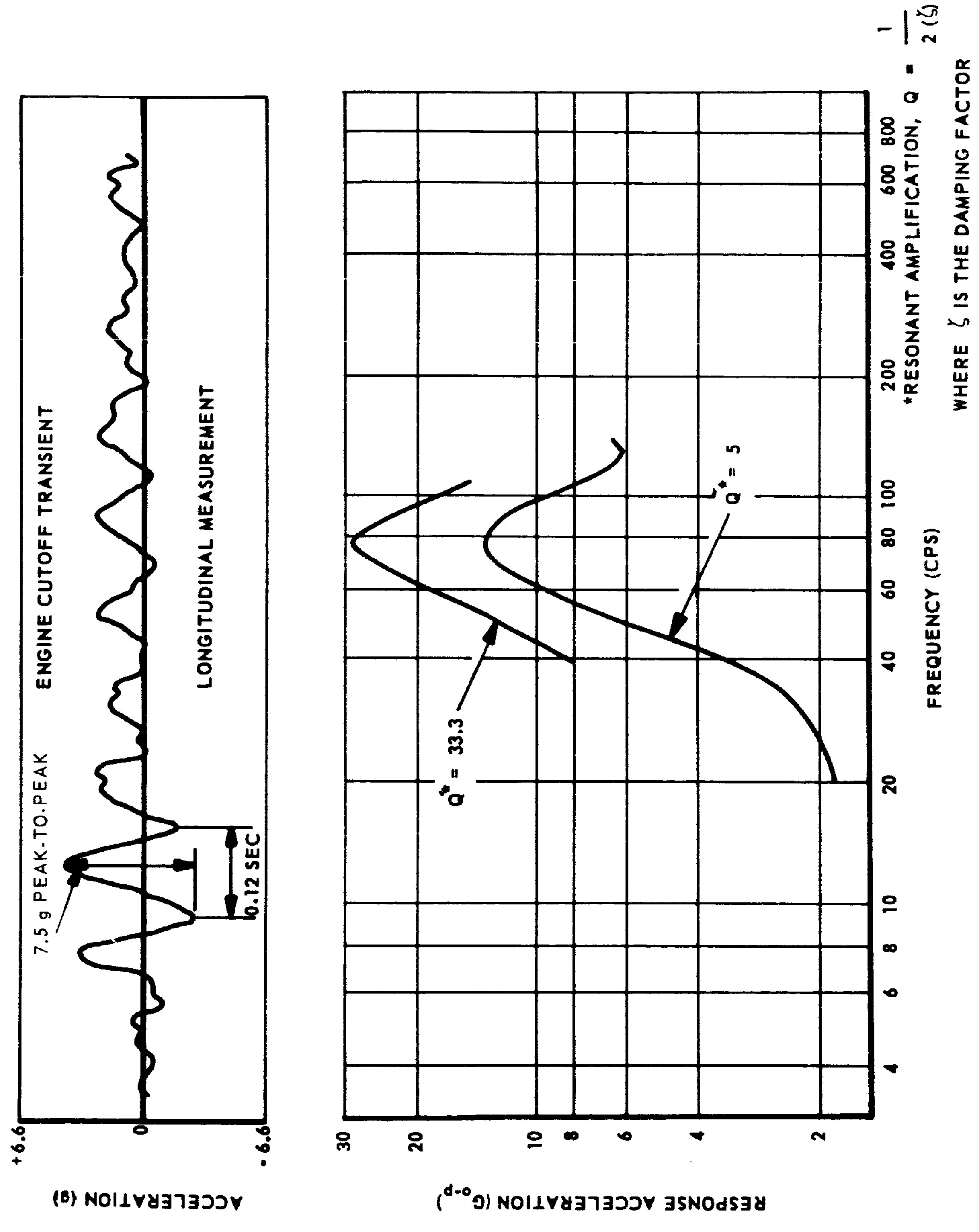

Figure 7-5 Shock Spectra of Agena Engine Cutoff, Aft Equipment Rack, Station 424

Table 7-3

LONGITUDINAL SINUSOIDAL VIBRATION ENVELOPE

(TAT, Thor, and Atlas)

Equipment*		Structure**			
		Forward Rack		Aft Rack	
Frequency (cps)	Magnitude (0-peak)	Frequency (cps)	Magnitude (0-peak)	Frequency (cps)	Magnitude (0-peak)
5 — 17	0.25 in	5 — 9	0.25 in	5 — 9	0.25 in
17 — 22	7.0g	9 — 16	2.0g	9 — 16	2.0g
22 — 40	5.0g	16 — 22	4.0g	16 — 22	3.0g
40 — 400	7.5g	22 — 100	2.0g	22 — 100	2.0g
400 — 3000	20.0g	100 — 200	4.0g	100 — 250	3.0g
		200 — 400	5.0g	250 — 400	5.0g

*The equipment test levels are for equipment weighing 75 lb or less and represents an envelope of the maximum levels specified for different locations within the Agena. Equipment levels are described in greater detail in LMSC-6117D.

**Values are for the total rack structures respectively.

Table 7-4

TRANSVERSE SINUSOIDAL VIBRATION ENVELOPE

Equipment*		Structure**			
		Forward Rack		Aft Rack	
Frequency (cps)	Magnitude (0-peak)	Frequency (cps)	Magnitude (0-peak)	Frequency (cps)	Magnitude (0-peak)
5 — 17	0.25 in	5 — 6.5	0.25 in	5 — 6.5	0.25 in
17 — 22	7.0g	6.5 — 100	1.0g	6.5 — 100	1.0g
22 — 40	5.0g	100 — 200	2.0g	100 — 250	2.0g
40 — 400	7.5g	200 — 400	5.0g	250 — 400	4.0g
400 — 3000	20.0g				

*Same as Table 7-3.
**Same as Table 7-3.

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Table 7-5

LONGITUDINAL AND TRANSVERSE RANDOM VIBRATION ENVELOPE OF EQUIPMENT*

Frequency (cps)	Spectral Density (g^2/cps)	Overall Level (g rms)
20 — 400	0.20	26.8
400 — 2000	0.40	

Table 7-6

ACOUSTIC EXCITATION ENVELOPE

Equipment*		Structure (Forward Rack)**	
Frequency Band (cps)	Sound-Pressure Level (db)	Frequency Band (cps)	Sound-Pressure Level (db)
37.5 — 75	128	37.5 — 75	149
75 — 150	132	75 — 150	154
150 — 300	146	150 — 300	157
300 — 600	146	300 — 600	156
600 — 1200	147	600 — 1200	155
1200 — 2400	148	1200 — 2400	153
2400 — 4800	148	2400 — 4800	150
4800 — 9600	148	4800 — 9600	147
Overall Level	154	Overall Level	163

*Same as Table 7-3.
**Values are for total rack structure.

Table 7-7

LONGITUDINAL AND TRANSVERSE SHOCK ENVELOPE OF EQUIPMENT*

Pulse Shape	Magnitude (0-peak)	Time (sec)
Half Sine Wave	40g	0.008

*Same as Table 7-3.

Table 7-8

QUALIFICATION TEST TIME

Excitation	Equipment	Structure
Sinusoidal	Sweep at 3 minutes per octave along each of 3 mutually perpendicular axes	Sweep at 3 minutes per octave along each of 3 mutually perpendicular axes
Random	5 minutes in each of 3 mutually perpendicular axes	
Acoustic	5 minutes	5 minutes

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SECTION 8
THERMAL CRITERIA

8.1 GENERAL

This section summarizes the thermal environment criteria that are imposed on the spacecraft by the Agena, nose shroud, and spacecraft adapter during prelaunch, liftoff, and ascent. Criteria are presented as an envelope of values for typical missions employing Atlas, Thor, and TAT boosters. This data was compiled from flight histories, tests, and analytical predictions.

During the design and development phase of a mission, ascent and coast phase temperature histories are calculated for selected temperature-critical spacecraft components. In the event that the calculated temperatures exceed the design limits of the spacecraft components, a change in trajectory, launch window, or spacecraft configuration is recommended to eliminate the thermal problem.

The temperature history of a spacecraft component is determined by the response of the spacecraft to heat rates incident on that component. Heat is exchanged between the surroundings and the spacecraft by any combination of the three possible methods of heat transfer; conduction, convection, and radiation. All three modes of heat transfer occur continuously while the spacecraft is on-pad, whereas only radiation and conduction will influence the component once shroud depressurization has occurred during the ascent phase of flight.

The ascent phase thermal environment is conveniently discussed by dividing it into two phases: shroud-on and shroud-off. During the shroud-on phase of flight, radiation exchange between the shroud inner surface and Agena diaphragm and the spacecraft is of prime interest. During the shroud-off phase of flight, radiation exchange between the earth, space, and various spacecraft components occurs with solar, albedo, and free-molecule flow heat rates incident on the exposed surfaces. Conduction between spacecraft components occurs at all times; the magnitude of heat exchange depends on the spacecraft design.

Thermodynamic analyses of Agena-boosted NASA spacecraft have been performed for Mariner Mars, S-27, Nimbus, EGO, and Comsat. Temperature histories of spacecraft components will not be presented in this document since the temperature response is dependent on the spacecraft design and each spacecraft will respond in a different manner. However, representative "backface" shroud temperatures (i.e., of areas facing the spacecraft) and heat rates which may be expected in space, are presented. Selected factors (spacecraft design, trajectory parameters, and launch window) influencing the temperature response of the spacecraft are discussed in the following paragraphs.

8.2 SHROUD-ON ASCENT

The temperature response of the spacecraft during the shroud-on ascent phase is determined by the particular spacecraft design and the inner shroud surface temperature exposed to the spacecraft. Shroud temperatures are dependent on ascent trajectory and shroud design, and vary as a function of the distance from the nose cap. Since these shroud surfaces will in most cases radiate excessive thermal energy to the spacecraft, additional thermodynamic protection is required. This protection is chosen to satisfy the requirements of the individual spacecraft and consists of either insulation or radiation shields. Radiation shields consist of polished aluminum (low emittance) surfaces mounted between the shroud surface and the spacecraft. Conduction between the shroud and the shield is minimized by using spacers that have a low thermal conductivity.

Contamination restraints are satisfied by designing the shroud so that all inner surfaces remain below their respective smoking temperatures. The

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two most widely used shroud structural materials are magnesium and fiberglass laminate. An indication of the thermal behavior of each is evident from Fig. 8-1, in which backface temperature histories are plotted for locations on the magnesium ogive portions of the Mariner Mars shroud and for a location on the Nimbus-type fiberglass shroud.

To properly evaluate the thermodynamic environment experienced by the payload, the effects of the first-stage booster must be considered. Therefore, a characteristic temperature curve is provided for (1) the Nimbus-type shroud with an Atlas, Thor, and TAT booster, and (2) the Mariner Mars shroud with the Atlas booster. The temperature histories presented are those that have been predicted for the missions specified and must be used only for general planning. The temperature histories for a particular shroud system must be calculated for the particular trajectory and desired protection.

The Nimbus-type shroud consists of fiberglass cylinder and cone sections with a fiberglass nose cap. An aluminum radiation shield is provided below the nose cap to shield the spacecraft from thermal radiation. Insulation with a nominal inside emittance of 0.85 is provided from the nose cap to the bottom of the shroud. The insulation thicknesses are determined for the unique mission requirements. A lower emittance could be provided through use of aluminum heat shields if RF transparency were not required. Figures 8-2 through 8-4 present typical temperature histories for no insulation and 1 inch thick insulation for an Atlas/Agena (EGO), Thor/Agena (Nimbus), and TAT/Agena (POGO). As can be seen, the highest temperatures predicted occur for the Atlas/Agena boost.

The inside surface temperature histories that affect the spacecraft for the Mariner Mars shroud and trajectory are presented in Figure 8-5. The Mariner Mars shroud consists of a beryllium nose cap, magnesium ogive section, and a magnesium, ring-stiffened shell section. The

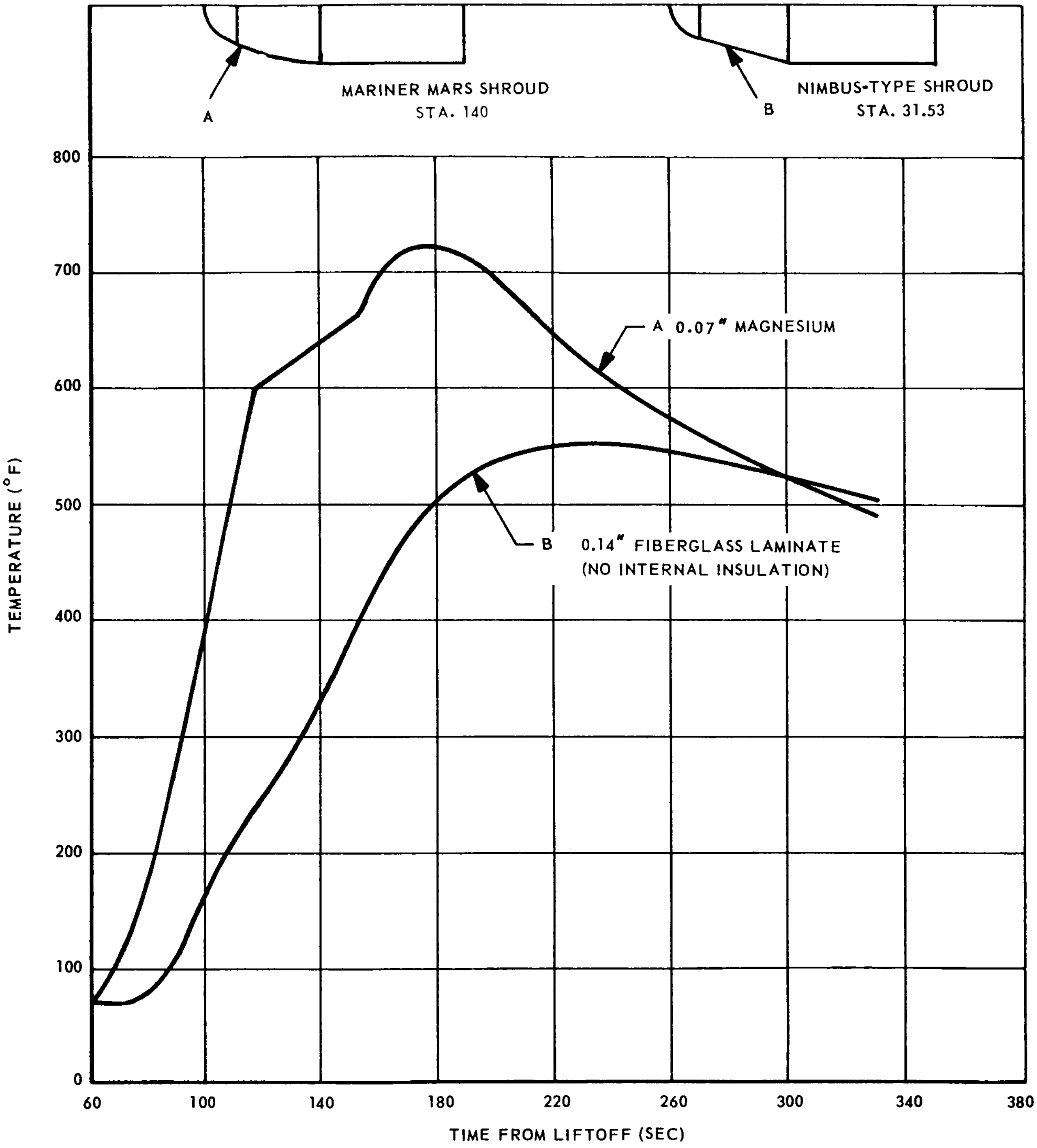

Figure 8-1 Typical Backface Temperature Histories for Three Shroud Materials

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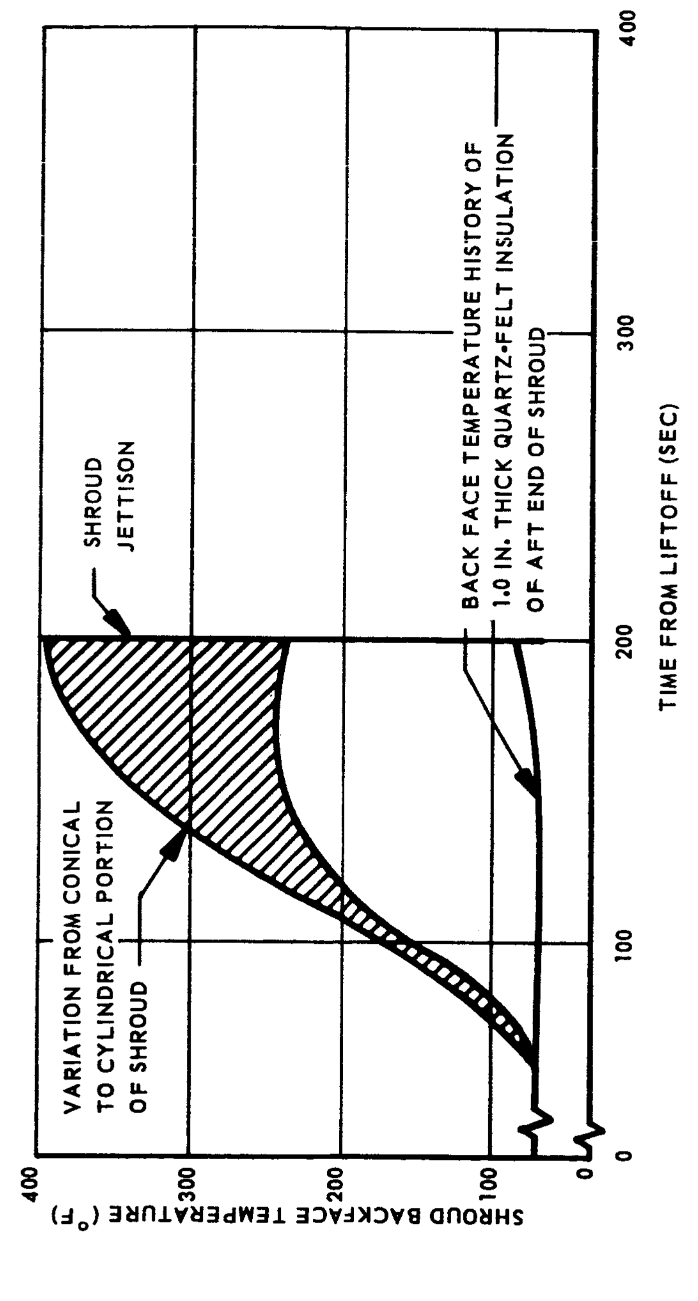

Figure 8-2 Shroud Backface Temperature History, Thor/Agena Missions, Nimbus-Type Shroud

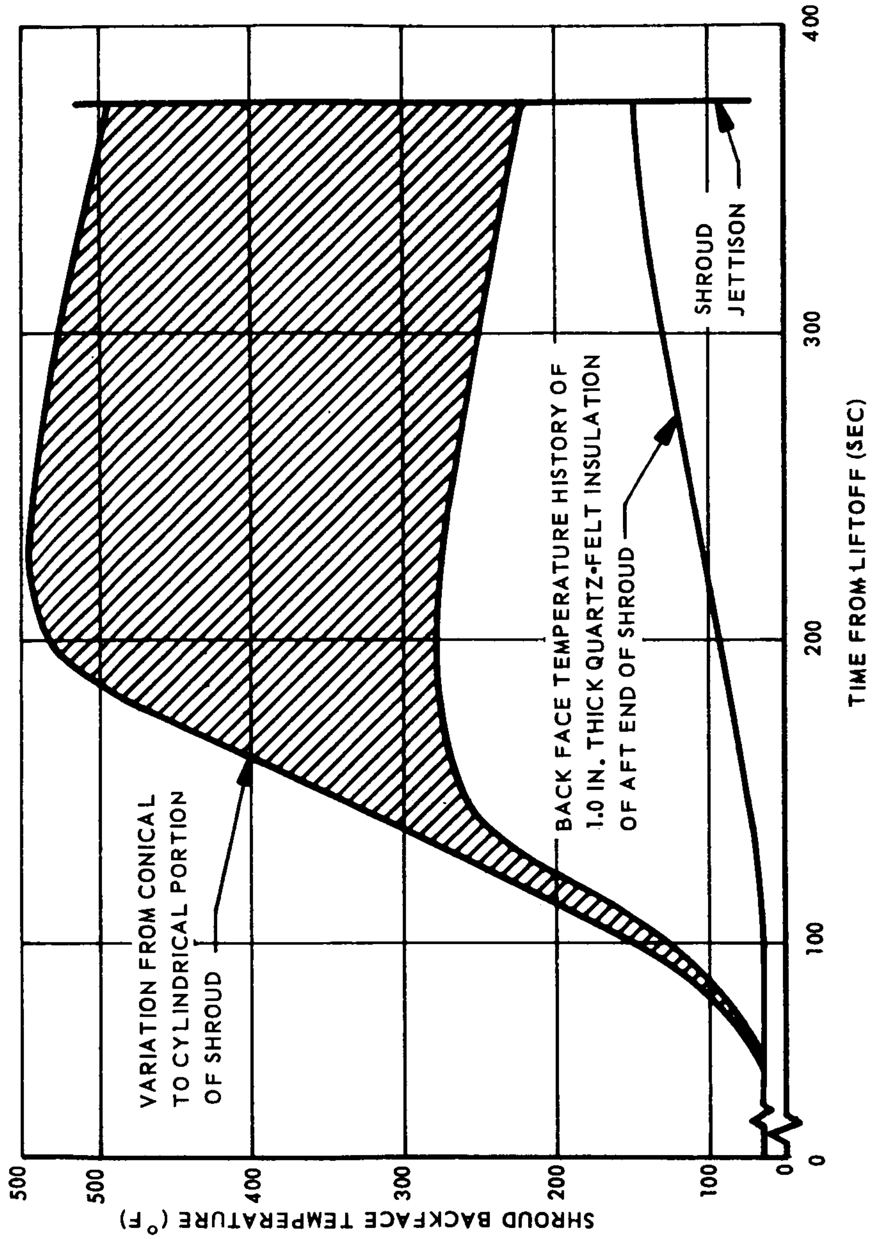

Figure 8-3 Shroud Backface Temperature History, Atlas/Agena Missions, Nimbus-Type Shroud

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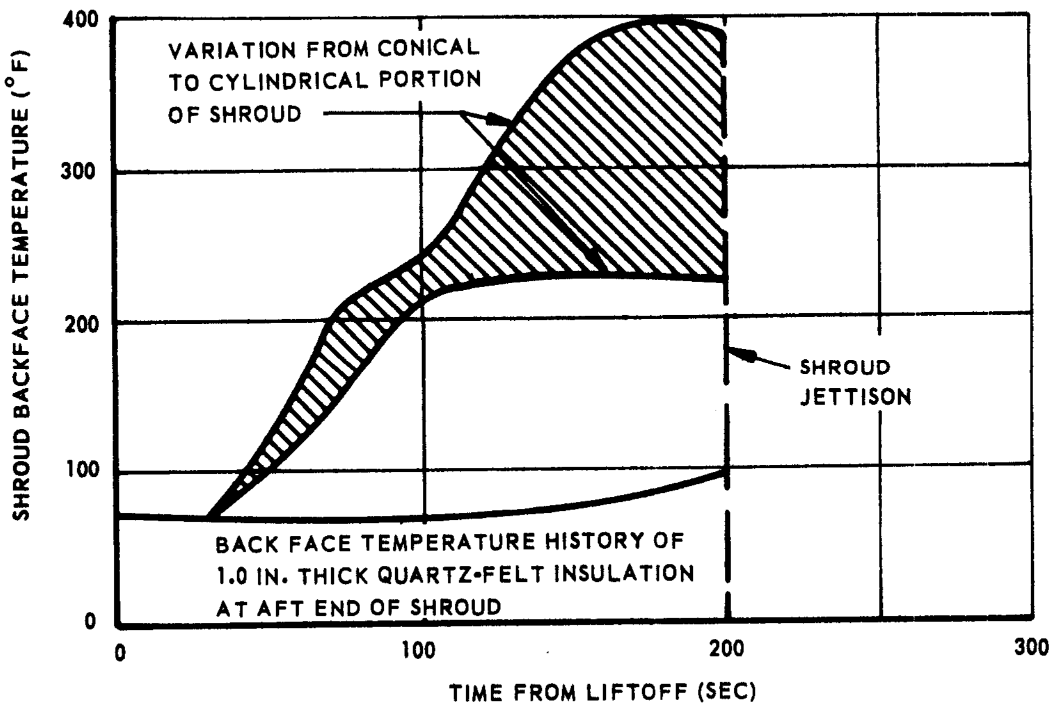

Figure 8-4 Shroud Backface Temperature History, TAT/Agena Missions, Nimbus-Type Shroud

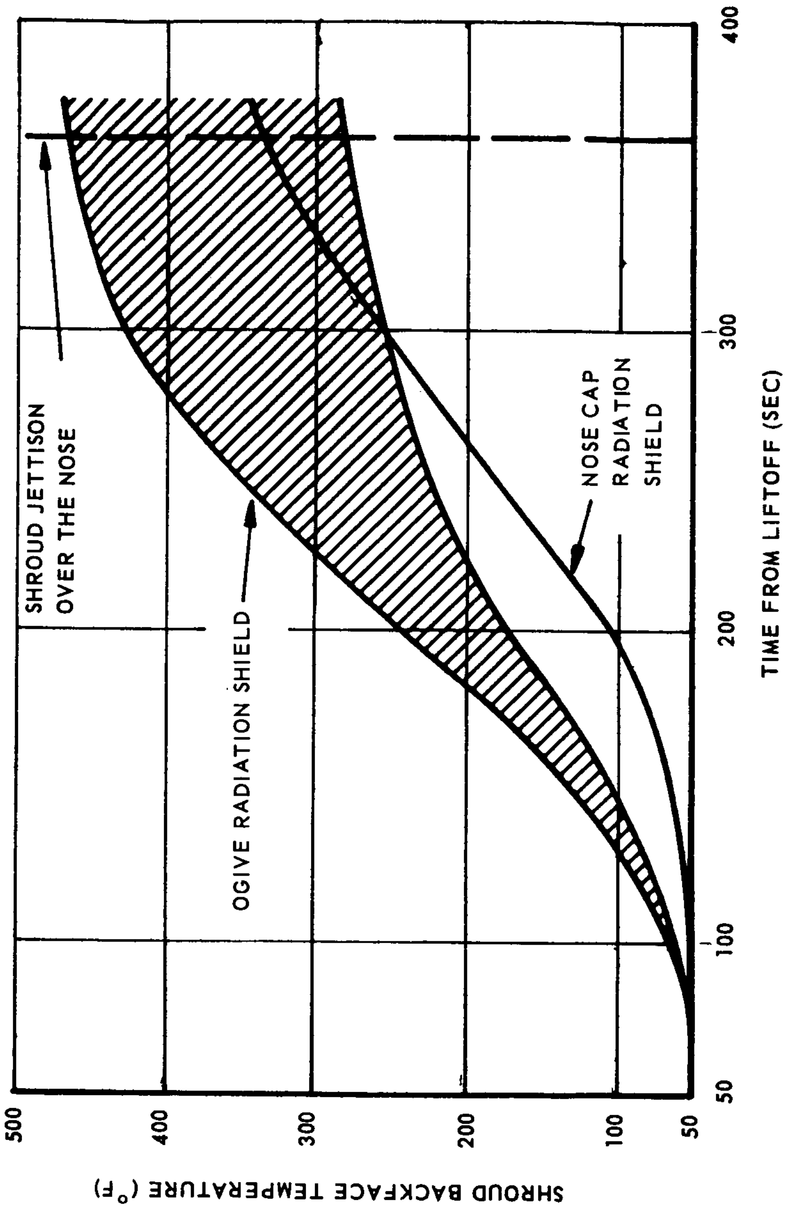

Figure 8-5 Shroud Radiation Shield Temperature History, Atlas/Agena Missions, Mariner Mars Shroud and Trajectory

thermal radiation from the beryllium nose cap is partially reflected back to the shroud by a double radiation shield. The temperature history of the inner of these two shields for Mariner Mars is presented in Fig. 8-5. This shield has a nominal emittance of 0.09. The temperature history of the ogive shield is also shown in Fig. 8-5. The ogive shield extends the length of the magnesium ogive section; its nominal emittance is 0.09. A temperature band is shown to indicate the temperature gradient along the shield. The maximum temperature occurring on the cylinderical shield was 240°F. This section also has a nominal emittance of 0.09.

8.3 SHROUD-OFF ASCENT

The period of flight from shroud jettison to spacecraft orbital injection is termed shroud-off ascent. This period is approximately 1 hour or longer for orbital missions and varies from 15 to 45 minutes for a probe mission, depending on the launch window. During this phase of flight, the payload is subjected to solar, albedo, earthshine, and free-molecule flow heat rates. Typical parallel and normal free-molecule flow heat rates are presented in Fig. 8-6. Note that the free-molecule flow heat rates may approximate the solar constant in magnitude. The solar, albedo, and earthshine heat rates incident on a payload component are determined by the launch date, launch time, trajectory flown including programmed attitude changes, orientation of the payload component, and shading of the component by another portion of the spacecraft or the Agena. After the heat rates have been determined and a thermal model is established for the spacecraft's temperature-critical components, the calculated heat rates are used as an input for a computer program. From the results of this program, temperature histories of the critical spacecraft components are determined for the desired launch window. If the temperature limits established by the spacecraft manufacturer are exceeded, recommendations for an increased Agena injection altitude or restriction of the launch window is suggested. Typical

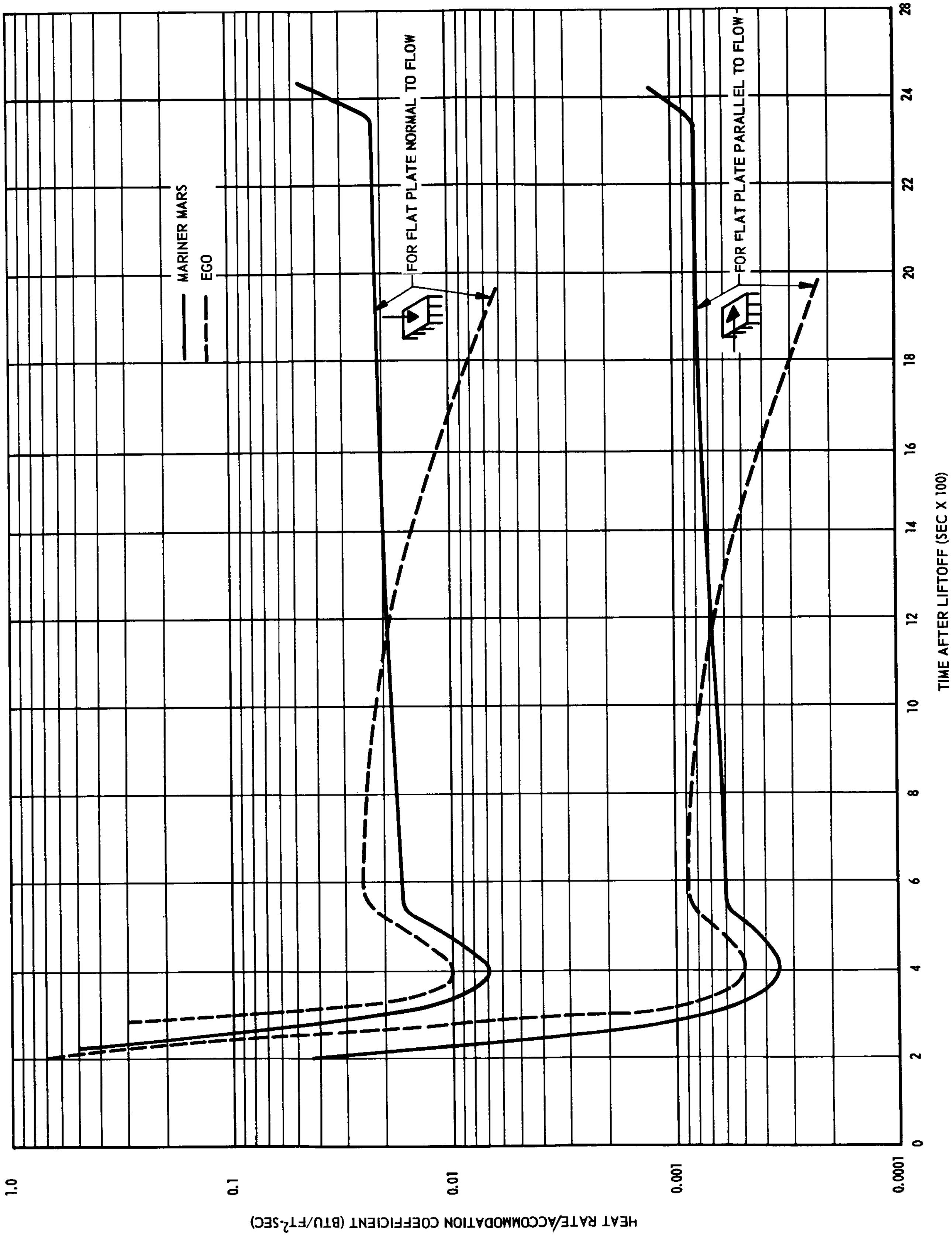

Figure 8-6 Free-Molecule Flow Heat Rates for the Mariner Mars and EGO Missions

temperature histories of spacecraft components are not shown in this document since the temperature response depends entirely on the particular trajectory, launch window, and the spacecraft design details.

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8.4 SPACECRAFT PRELAUNCH COOLING METHODS

8.4.1 Spacecraft Heat Sources

The on-pad conditioning requirements for the spacecraft are determined by LMSC by performing a heat balance using the heat rates incident on the outside shroud surface and the spacecraft power dissipation. Outside heat rates consist of direct and diffuse solar radiation, ground and sky radiation exchange, and convection to the ambient environment. The amount of heat load which has to be absorbed by the cooling air is dependent on the air conditioning system and shroud used. Of the total heat load, as much as 90 percent may be attributed to outside heat sources. Power dissipation of the spacecraft studied to date has ranged up to 200 watts. Computer programs have been used for the present missions to perform the outside heat rate calculation based on the launch base and date, and to perform the heat balance on the system to satisfy the spacecraft environment restraints.

8.4.2 Internal Cooling

The most efficient method of cooling the spacecraft during on-pad operation is to directly introduce cooling media to the spacecraft cavity (i.e., inside the shroud). The spacecraft then is cooled and heat is transferred away from it by convection. The Nimbus-type shroud and the short version (Comsat) use this principle. After the required environment is established by the spacecraft contractor, the inlet air requirement is determined using the computer programs outlined in par. 8.4.1. The inlet air requirement is based on the power dissipation and shroud configuration (i.e., shroud insulation thickness) required for that mission. Figures 8-7 and 8-8 present the environment provided for Comsat, Nimbus, and EGO vehicles. It is

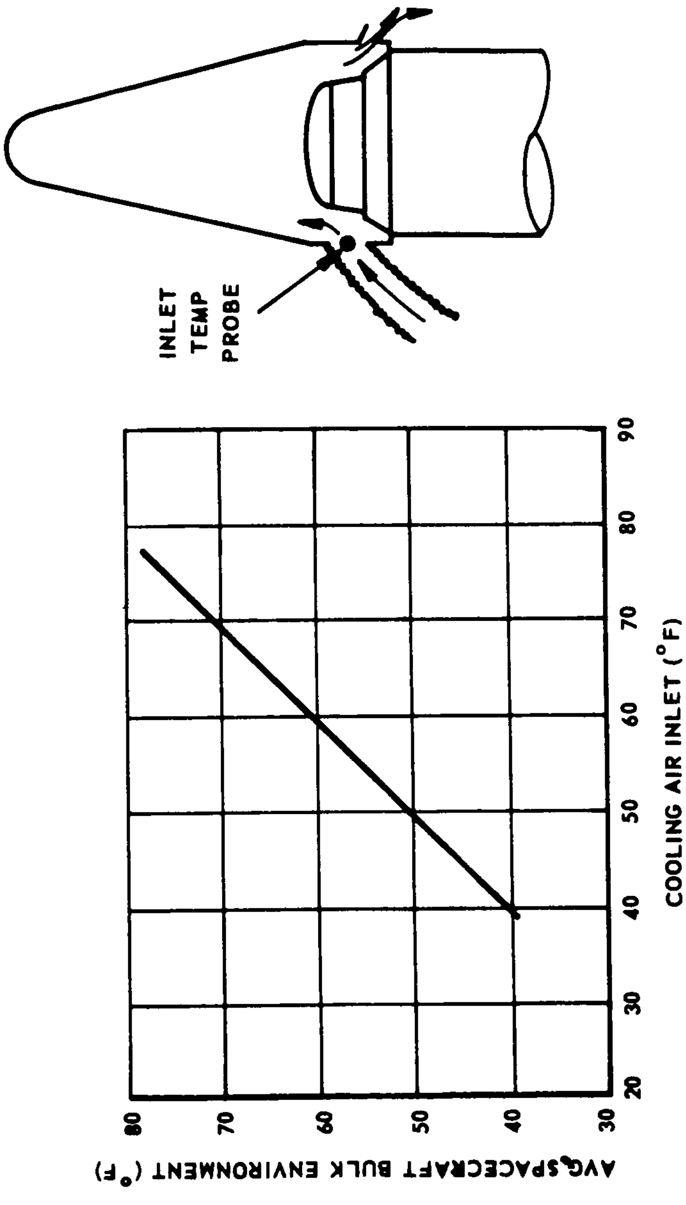

Fig. 8-7 Comsat Internal Cooling

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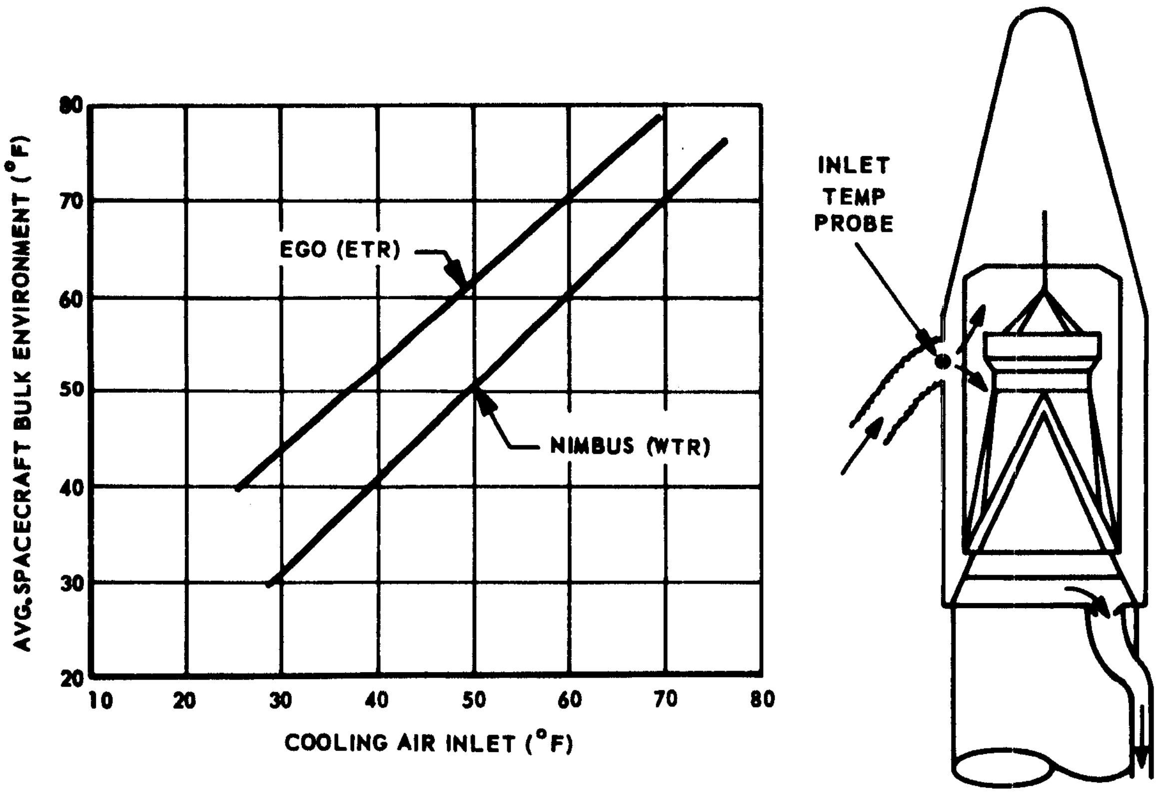

Fig. 8-8 Environment for Nimbus and EGO Internal Cooling at Specified Location

important to realize that these were determined based on the shroud insulation established for these programs and the power dissipation of these particular payloads. A payload of a different configuration and power level would result in a different environment. These environments are, however, representative of what may be expected. The limits of shroud inlet temperature shown are established by the equipment located at their respective pads. (Comsat and Nimbus, WTR 75-1-1; EGO, ETR 12). The environment shown for EGO was based on the temperature requirement at a specified location within the shroud, whereas the environments for Nimbus and Comsat are based on a "bulk environment." A bulk environment is defined as:

$$\frac{\text{Shroud inlet temperature + shroud outlet temperature}}{2}$$

The air-flow rate to maintain these requirements was 70 lb/min.

8.4.3 Contamination Control

An 1100 standard cubic feet per minute 0.3 micron air filter model, including a 10 micron prefilter, has been developed for installation on the inlet of the Type 15A air conditioner.

8.4.4 Internal Cooling by Other Systems

For Project Fire a combination of air and freon was used for spacecraft cooling, as shown in Fig. 8-9. Air cooling was used for general cooling and a small heat exchanger was used for concentrated heat loads. Temperatures from -200°F to $+120^{\circ}$F can be obtained by using a liquid freon refrigerant with the desired boiling point.

A nitrogen pre-cooling system, Figure 8-9, has been developed for a military program. In this system the nitrogen gas generated from liquid nitrogen was used to cool one component to lower than -40°F without excessively cooling other portions of the spacecraft.

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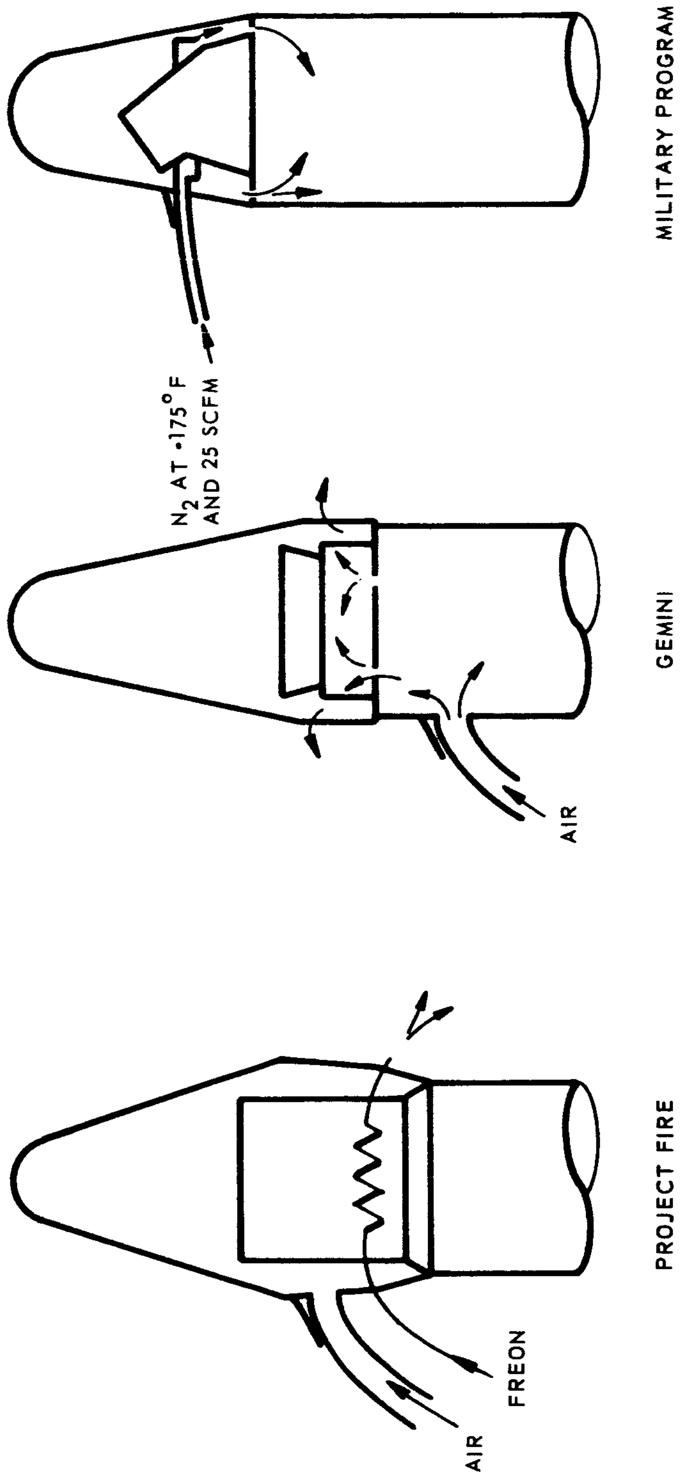

Figure 8-9 Alternate On-Pad Spacecraft Cooling Systems

8.4.5 External Cooling

Another method of cooling the spacecraft during on-pad operation is to place an external cooling source on the exterior of the shroud, thereby cooling the outer and inner surface of the shroud and relying on free convection within the cavity to cool the spacecraft. This method has been successfully employed on Ranger and Mariner Mars. External cooling is particularly useful for a spacecraft which requires stringent cleanliness. The shroud environment for any particular mission will be dependent on the launch date, power dissipation of the spacecraft, and shroud configuration including required radiation shields and shroud material and thickness. The Mariner Mars program used an externally mounted rubberized blanket. These systems have a 70 lb/min flow rate circulating through their passages and around the shroud. These external cooling methods are discussed further in Section 18.

The temperature control blanket is a vinyl-coated, nylon jacket shaped to fit snugly around the shroud, where it is fastened with a parachute-type, quick-release lanyard attached to the umbilical mast on the launch pad. During prelaunch preparations horizontal and vertical zippers can be unfastened to expose access doors in the shroud. Umbilical cable tension at vehicle liftoff releases the blanket, which falls away from the shroud.

The Mariner Mars cooling blanket inlet temperature required for maintaining a specific spacecraft environment temperature is shown in Fig. 8-11. Similar blankets have been used for the Vela Satellite and Ranger programs. The performance of the blanket is predicted on the Mariner Mars mission; the requirements of another mission may result in a different performance.

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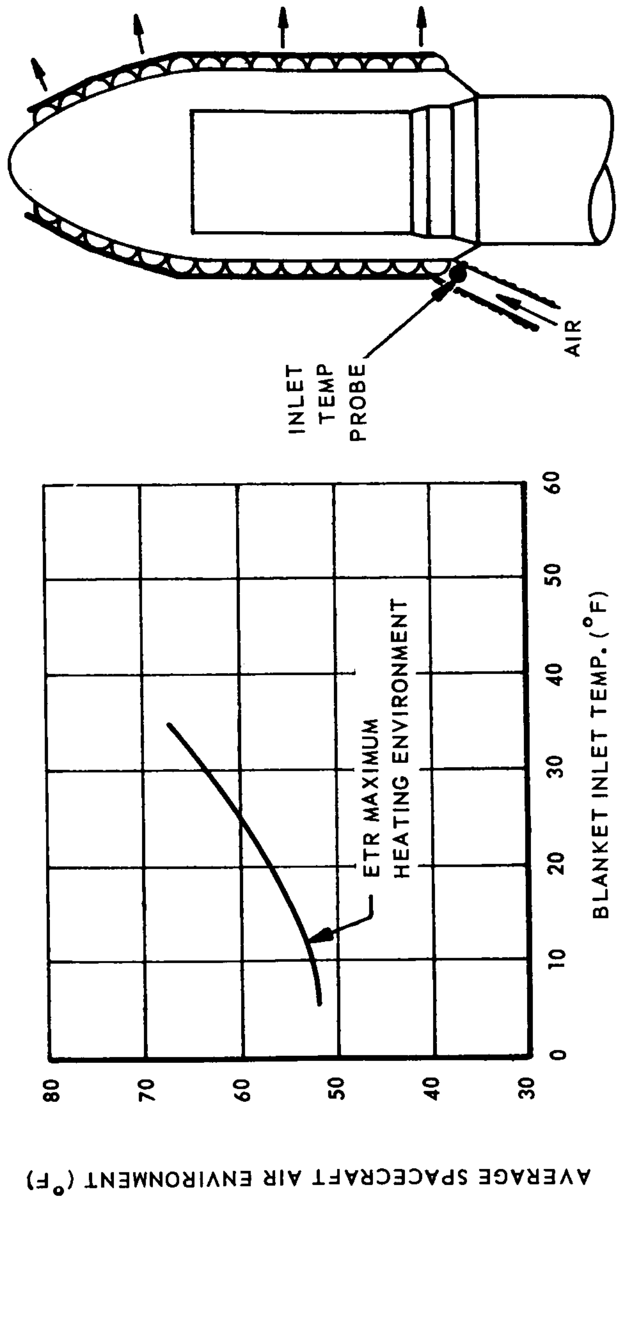

Figure 8-11 Mariner Mars Cooling Blanket Inlet Temperature Requirement

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SECTION 9
MISCELLANEOUS ENVIRONMENTAL CRITERIA

9.1 GENERAL

This section summarizes the miscellaneous environmental criteria imposed on the spacecraft by the launch vehicle. Included are interface material limitations, electromagnetic interference and nuclear radiation. "General Environmental Specification for Equipment of the Agena and Associated Payload," LMSC-6117D, provides additional environmental criteria which can be utilized as a guide by the spacecraft designer.

9.2 INTERFACE MATERIAL LIMITATIONS

This paragraph summarizes the material considerations pertinent to the spacecraft caused by contact or close association with the Agena vehicle during testing, handling, prelaunch, and ascent. Table 9-1 summarizes the principal phenomena to be considered. It covers only Agena boost-type missions which impose environments usually not exceeding one hour. Long-term conditions such as solar and other penetrating radiation, meteoritic impact and erosion, as well as other near-space environments that are significant during Agena orbital missions are not presented.

9.3 ELECTROMAGNETIC INTERFERENCE (EMI)

Interference produced within the vehicle system shall be controlled to prevent undesired interaction and malfunctioning of all other electronic and electrical subsystems regardless of whether the subsystem is electrical, aural, video, or mechanical. The spacecraft designer must consider the radiated and conducted interference generated by the Agena vehicle as well as the susceptibility of the Agena to interferences generated external to

Table 9-1

MATERIAL CONSIDERATIONS RELATIVE TO ENVIRONMENTAL EFFECTS

Period	Conditions	Phenomena	Where	Limit	Comments
Preflight- (during mating, testing, and handling)	Atmospheric dust and moisture	Ordinary Corrosion	Agena	No	Agena materials selected for corrosion resistance sufficient for normal periods of storage and activity
		Galvanic Corrosion	Interface	Yes	Mating surface materials on spacecraft side of interface should be selected of magnesium alloys, aluminum alloys (preferably clad), or others close to magnesium in electromotive series. If this is not feasible, electrodeposited coatings, paints, organic finishes and/or other means should be taken to prevent a galvanic couple. (Mariner Mars V-band shoes and mating pads are magnesium ZK60A; Ranger interface ring is magnesium thorium alloy, HM-21A-T5)
Ascent (between launch and shroud ejection)	Rapid pressure drop to near vacuum value plus temperature rise due to aerodynamic heating of shroud	Metallic material evaporation	Shroud cavity	No	Eliminated cadmium plating from hardware exposed to environment inside the shroud. Also used silver-plated fittings and titanium fastener materials
		Nonmetallic material smoking for outgassing	Shroud cavity	No	Outgassing minimized by proper selection of heat resistant materials, post-cure of fiberglass laminates, silicone rubber seals around openings etc. Spacecraft isolation diaphragm and inert gas purge for some missions prevents contamination from forward rack area.
		Loss of lubricants	Agena and shroud cavity	No	Molybdenum disulfide and other dry lubricants resistant to vacuum conditions used
		Thermal expansion/ contraction stresses	Interface	Yes	Materials and design tolerances should be selected for the spacecraft side of the interface to permit growth or contraction presented by worst combination of component temperature differences.
Ascent (between shroud ejection and spacecraft ejection)	Near-vacuum, large temperature variation between sunlit and shaded sides (ranging from cryogenic to elevated temperatures)	Seizing due to high vacuum	Interface	Yes	Mating surfaces on spacecraft side of interface should be machined to a surface finish of at least 63 smoothness, and mating will be accomplished in the dry condition.
		Low temp. embrittlement	Agena	No	Materials selected for Agena with sufficient toughness to function at low temperatures.
		Thermal expansion/ contraction stresses	Interface	Yes	Space environments present even more extreme temperature variations than while the shroud is on. (Same comments as for this phenomenon above.)

Agena, e.g. interference from AGE, launch and tracking facilities, spacecraft, etc. The noise levels generated by the Agena are strongly dependent on the configuration for the particular mission and the designer must keep this in mind when designing the spacecraft for EMI control.

A partial list of precautions to be followed for the purpose of minimizing EMI problems across the spacecraft/Agena electrical interface are presented below. "Agena Systems Electrical Interface Specification", LMSC-447969, which is derived from MIL-I-26600, lists all the requirements for EMI control.

a. For electrical continuity and isolation of circuits, electrical shields shall be maintained through the interface connector. All audio frequency (AF) (0 to 150 Kc) telemetry or power signal shields shall be grounded at one end only, preferably at the signal source. Substitute method is to float the shield at the interface connector and ground the shield on both sides of the interface at their respective ground points. Shields of interconnecting wire shall not be grounded at more than one point nor be connected in any way which creates a loop having nominally zero ohms impedance.

b. For each interface circuit the signal and its return shall be routed together in the same twisted pair (triple twisted wires for three phase power). Exception to the foregoing are the wires that carry telemetry signals to the Agena telemetry system where a common signal return is utilized, i.e., commutated data.

c. Transducers telemetered by the Agena shall be energized by power supply which is common to the Agena telemetry system.

d. It is mandatory that a transient suppression device be used in every case where direct current that flows through an inductive load is interrupted (e.g., relay coil). The diode or other suppression device shall be located as close to the inductance as possible.

e. All pyrotechnic power shall be conducted through shielded twisted pairs with the shield grounded at both the source and load ends. The pyrotechnic device enclosure shall be properly bonded to vehicle structure.

f. In compliance with general range safety plan AFMTCP 80-2, the electro-explosive device (EED) no-fire current and no-fire power shall not be less than 1 ampere and less than 1 watt, respectively, as the result of the application of a direct current for five minutes. In lieu of the foregoing, the range user may validate the survival of each EED before, during, and after installation in the following electromagnetic fields:

Frequency Range	Field Intensity
150 Kc up to and including 50 Mc	2 watts per square meter (28v per meter)
above 50 Mc	100 watts per square meter (194 volts per meter)

g. Equipment containing electrical circuits and pyrotechnic devices which may be susceptible to RF or produce RF, shall be so installed that there will be a continuous low impedance path from the equipment enclosure to the vehicle structure.

h. The spacecraft shall be electrically bonded to the spacecraft adapter and the spacecraft adapter shall be electrically bonded to the Agena structure.

i. It is recommended that any sensitive digital type device in the payload have its own isolated power supply separate from the Agena supply.

j. For programs using Bell Telephone Laboratory (BTL) guidance installed in the Agena forward rack, the maximum magnetic field from any external source shall not exceed 3 gauss at the surface of the guidance set.

Component EMI tests have been recently initiated at LMSC to determine the noise and frequency level of interference emanating from individual "black boxes" and susceptibility of these to conducted, radiated, and intermodulated interference signals. These tests by themselves are not conclusive when evaluating subsystems and system compatibility. Depending on the nature of the spacecraft and the mission, it may be desirable that the spacecraft, Agena, and booster contractors jointly conduct EMI and RF compatibility analysis and test programs for the purpose of eliminating EMI and RF incompatibilities that may exist between various segments of a total space system.

9.4 NUCLEAR RADIATION ENVIRONMENT

The Agena has no components which exhibit extreme sensitivity to a nuclear radiation environment. The only items of a radiation sensitive nature are the electronics and guidance and control equipment located in the forward equipment rack.

Spacecraft employing nuclear reactors or radioisotope components may require that certain Agena components be shielded or that radiation hardened components be employed on the Agena. Equipment that is capable of operating in a nuclear radiation environment has been developed on the SNAP program, which uses the Agena as both the boost vehicle and orbiting spacecraft for the testing of a nuclear reactor electrical power supply. This equipment includes regulators, batteries, inverters, relays, PAM telemetry, and certain mission peculiar equipment.

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SECTION 10
SHROUDS

10.1 GENERAL

Several shroud systems are available for use with the Agena D that provide spacecraft protection prior to launch and during ascent. Each shroud system has attachment and separation provisions, and spacecraft access or umbilical connector openings. These shroud systems have already been developed and are identified by the program name of application. The principal types are: Lunar Orbiter (Mariner-Mars type), Ranger, Nimbus, and OAO. General application information for each type is presented in the following paragraphs. Flight clearance envelope information is provided for those shrouds whose original mission applications necessitated evaluation in that area.

Presented also is the Standard Agena Clamshell Shroud, which is an improvement of the Agena Long Shroud (Nimbus and OGO type). This SAC shroud is intended for use in all future NASA Agena missions requiring a clamshell-type shroud of its envelope size and is expected to be available from shroud qualification tests scheduled for approximately the first quarter of the calendar year 1966.

10.2 LUNAR ORBITER SHROUD

The Lunar Orbiter shroud system is an over-the-nose type that includes a jettisonable shroud and separation systems. A transition section and diaphragm are supplied separately. This shroud is a one-piece structure, consisting of a magnesium barrel section, a magnesium conical and ogive section, and a beryllium nose cap. The transition section, which adaptes the 65-inch diameter shroud to the 60-inch diameter vehicle, is a ring-stiffened, magnesium cone attached to the adapter interface ring at Station 247.

The shroud is mounted over the spacecraft and is attached to the transition section at Station 242.25 by a V-band clamp. Release of the V-band clamp at shroud separation is provided by two pyrotechnic release fittings. Separation thrust is imparted to the shroud by four spring assemblies which develop approximately seven ft/sec relative velocity between the shroud and Agena/spacecraft.

The space enclosed by the shroud system measures 143.89 inches in length with a maximum diameter of 63.5 inches. (Diameter is constant from the base for a distance of approximately 89 inches and then tapers gradually to 24 inches at the forward end). Gross volume of this shroud is approximately 186 cubic feet. The shroud weighs 355 lb, the V-band clamp system weighs 19 lb, and the transition section, including the Agena/spacecraft diaphragm, weighs 37 lbs. Total drop weight is 374 lb. Overall shroud dimensions and details are shown in Figs. 10-1 and 10-2.

The forward dome is provided with a double element temperature radiation shield consisting of a 0.010-inch thick titanium liner and a 0.020-inch thick aluminum liner separated by a 1.50-inch air space. The ogive section contains an inner thermal shield of 0.020-inch thick aluminum formed by petals which allow for differential expansion. The barrel and conical sections radiation shields are made of a 0.016-inch thick aluminum in the same petal configuration, with an interlocking arrangement as shown in Fig. 10-1.

An air conditioning vent exhaust door is supplied in the shroud as shown in Fig. 10-1. The vent door provides exhaust area for the conditioning air which is used for temperature control of the spacecraft environment.

The flight clearance envelope and the shroud separation envelope are discussed in the following paragraphs for the Mariner Mars shroud and Mariner Mars spacecraft.

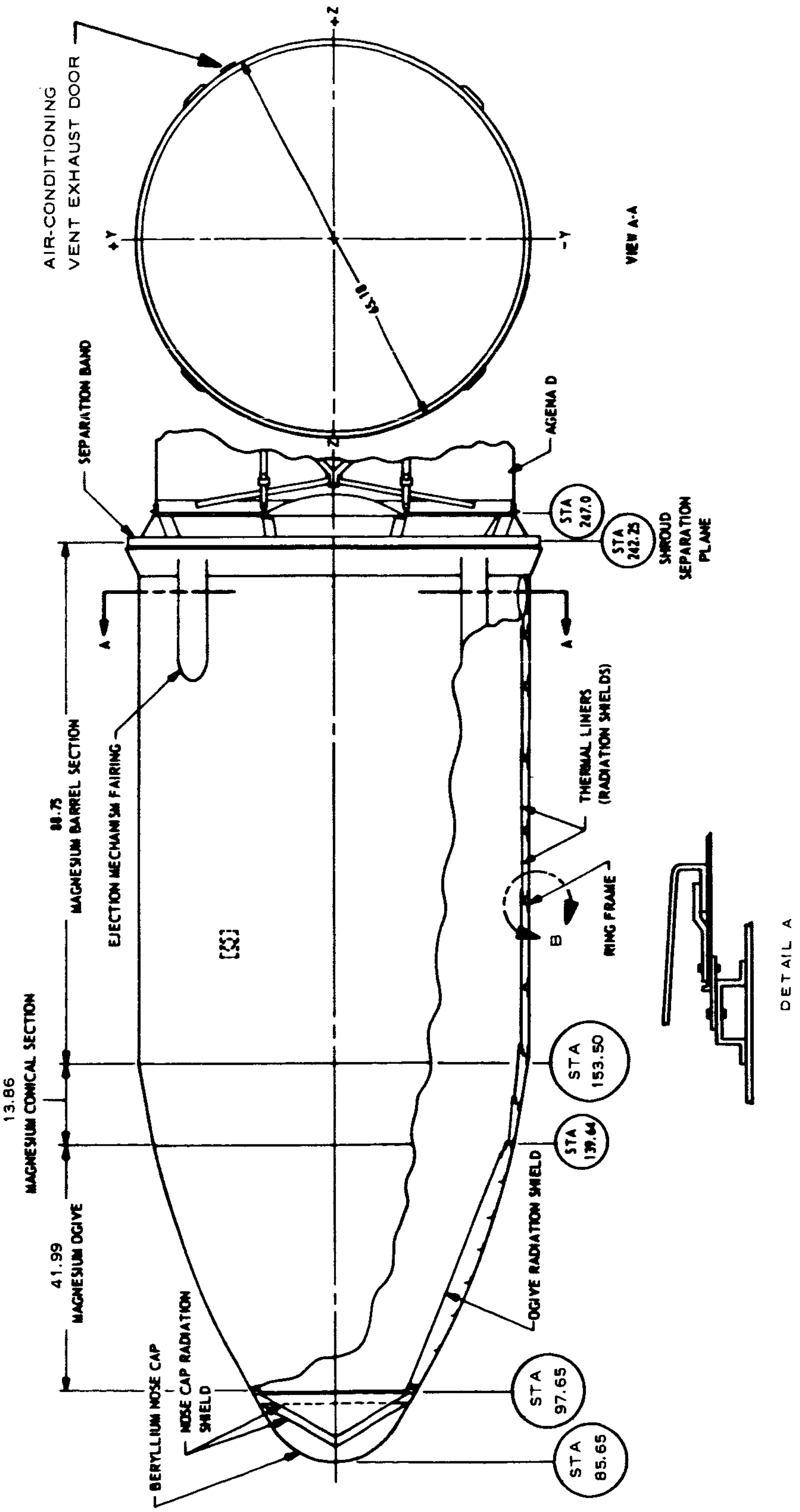


Fig. 10-1 Lunar Orbiter Shroud System

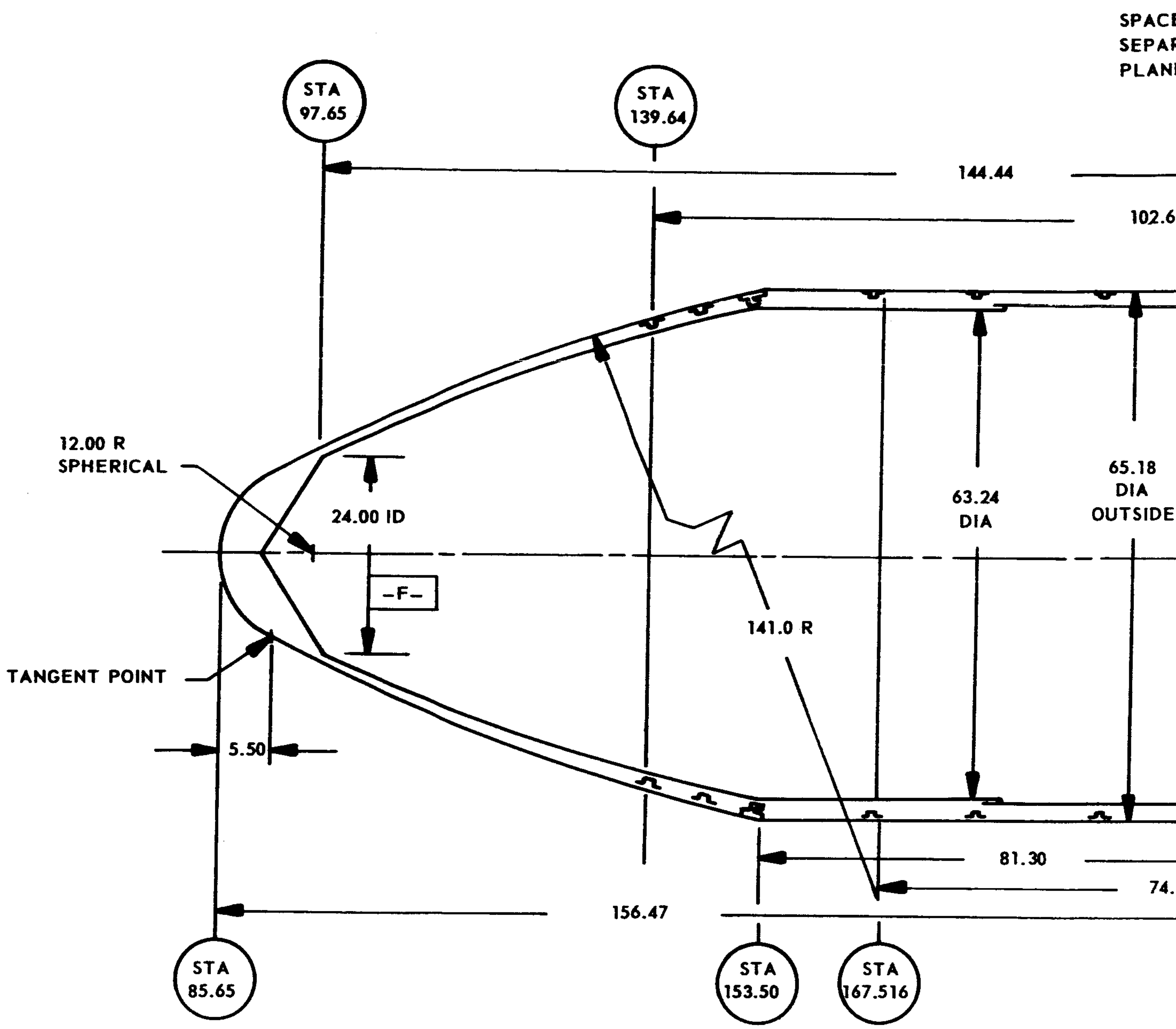
10.2.1 Flight Clearance Envelope

Radial deflections during ascent have been studied to determine that no impact between the spacecraft and shroud will occur. Mariner Mars spacecraft excursions, as well as the shroud deflections, are considered in establishing the total clearance envelope required. The calculation of the flight envelope for the Mariner Mars shroud and spacecraft is in three parts; manufacturing tolerances, vibration dynamics, and the air load bending induced during ascent to approximately 60,000 feet.

The manufacturing tolerance used in the case of the Mariner Mars shroud is 0.25-in. on diameter, or 0.125-in. on radius. This tolerance is considered to be constant through the full length of the shroud, including the ogive section and beryllium nose cap, and can be considered constant for any application of the Mariner Mars shroud.

The vibration excursion of various points on the shroud are calculated values and are presented as radial deflections at the stations listed in Fig. 10-3. These radial excursions of the shroud will vary with the weight and dynamic characteristics of the spacecraft because of the dynamic interaction of the two systems at the common Agena attachment plane. For this reason, values presented may be used only as a guide, exact values must be determined for each payload application involving spacecraft/ shroud clearances.

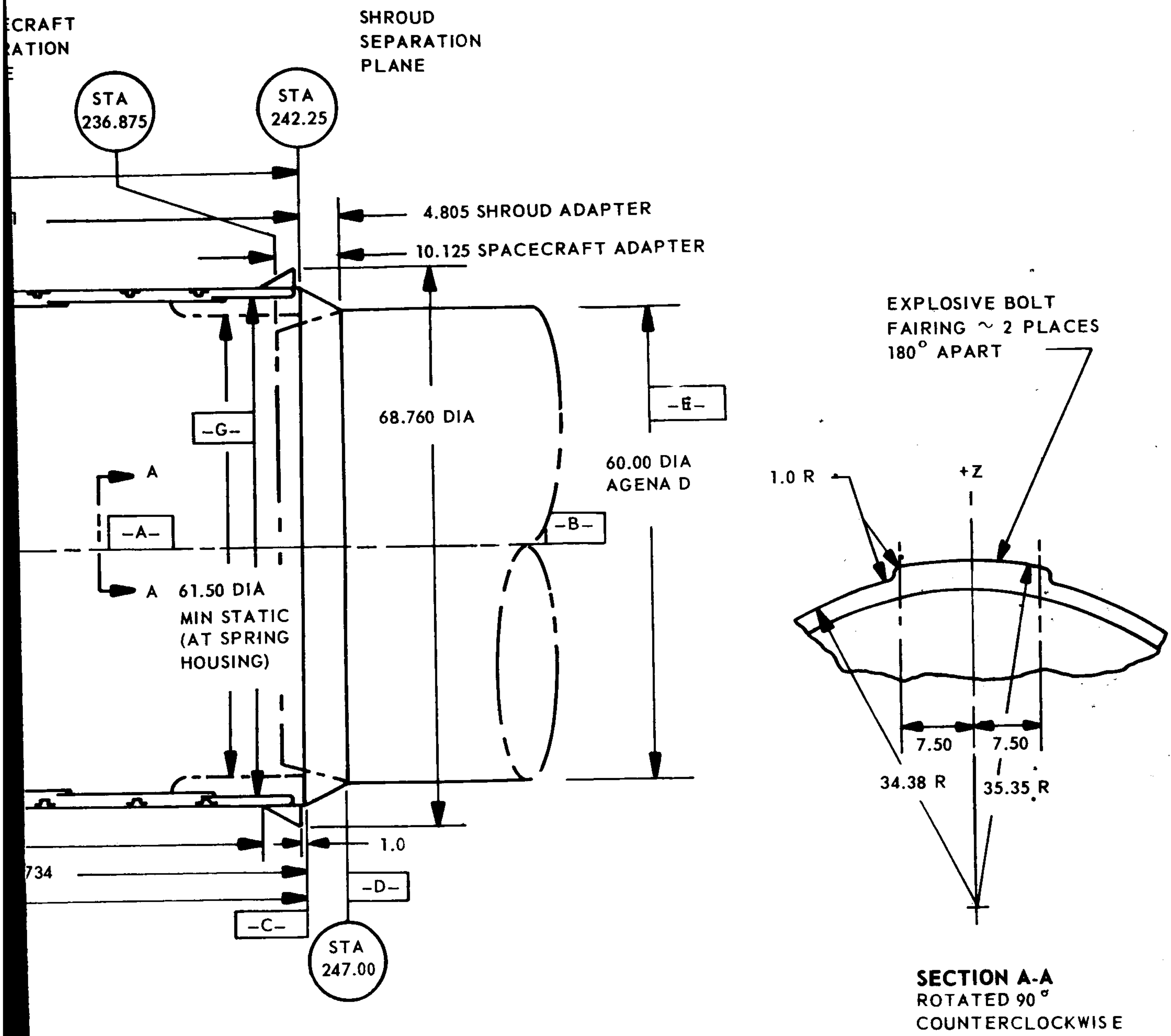
Bending loads are calculated values based on flight aerodynamic pressures predicted for the Mariner Mars ascent profile. Radial deflections due to flight bending loads are also presented in Fig. 10-3. Bending (static) deflections, like the vibration excursions, are mission-sensitive and will vary with the trajectory and booster system employed on the mission. The total predicted (maximum) radial excursion of the Mariner Mars shroud when used on the Mariner Mars mission can be obtained by adding the radial manufacturing tolerance, the radial vibration excursion, and the radial static deflection presented in Fig. 10-3.



BLOCK LETTER	DESIGNATION	STATION LMSC	TOLERANCES	DIAMETER
E	MOUNTING SURFACE OF AGENA	247.00	⊥ B .020 TIR	60.00
D	MOUNTING SURFACE OF SPACECRAFT-SHROUD ADAPTER	247.00	⊥ A .005 TIR	60.00
C	MOUNTING SURFACE OF SHROUD	242.25	∥ D .005 TIR	65.880
C	MOUNTING SURFACE OF SHROUD	242.25	⊥ A .005 TIR	65.880
G	INSIDE DIA OF SHROUD BASE	242.25	○ .12 TIR	65.338
C	MOUNTING SURFACE AT SHROUD BASE	242.25	⌒ .010 TIR	65.880
F	ID OF FORWARD HEAT SHIELD	97.65	⊙ C .150 TIR	24.00

-A-
-B-
TIR
⊥
∥

/



CENTERLINE OF SHROUD AND ADAPTER

CENTERLINE OF AGENA D

TOTAL INDICATOR READING

PERPENDICULARITY

PARALLELISM

○ ROUNDNESS

⌒ FLATNESS

⊙ CONCENTRICITY

Fig. 10-2 Lunar Orbiter Shroud System Overall Dimensions and Tolerances

2

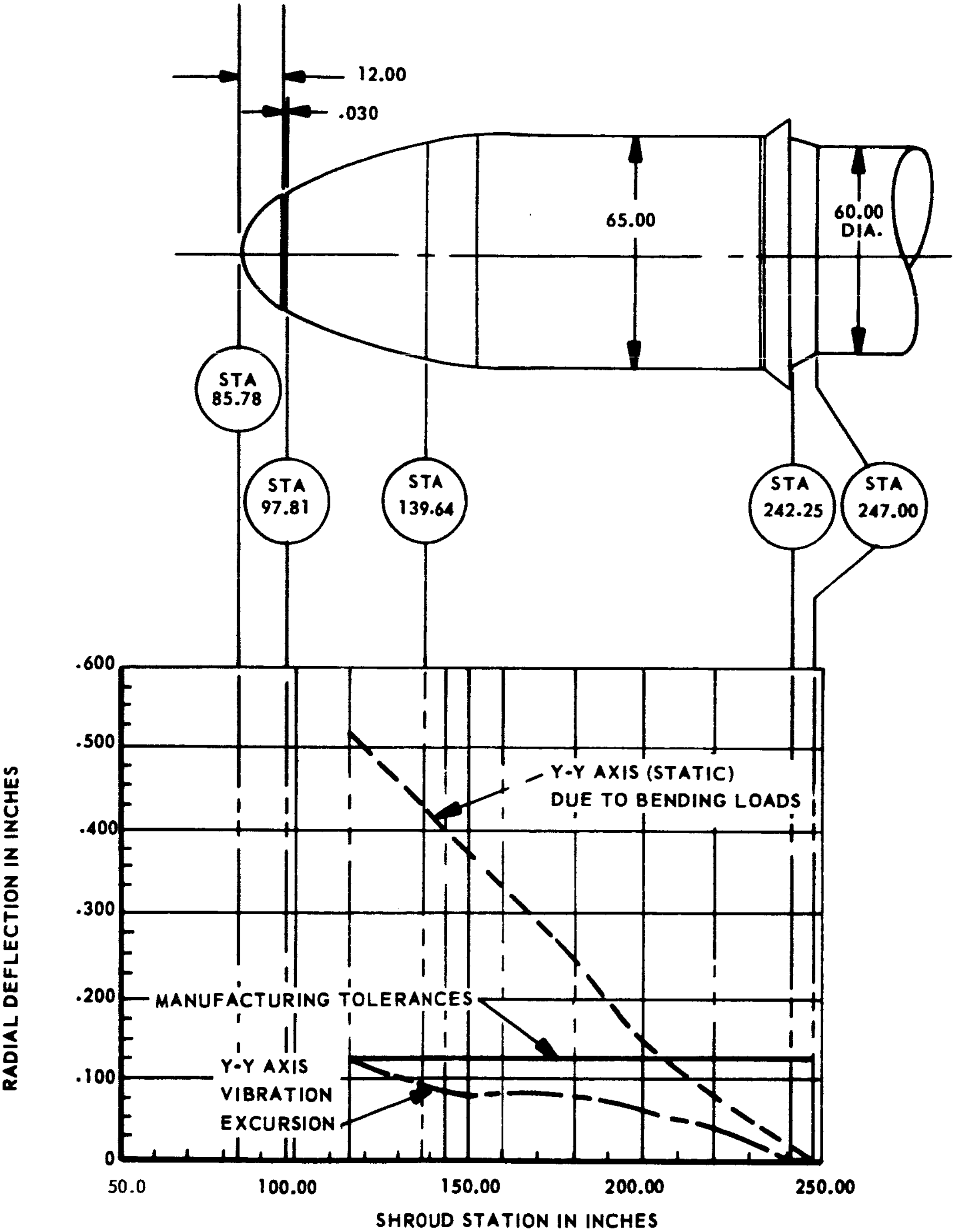


Figure 10-3 Mariner Mars Flight Clearance Envelope

10.2.2 Shroud Separation Envelope

The Mariner-Mars shroud is separated over the nose by the action of four ejection spring cartridge thrusters. The relative velocity imparted is approximately 7 ft/sec. The following three figures show the net reduction in shroud/spacecraft clearance as separation progresses, based on "worst condition" deviations of the various parameters, such as maximum - minimum separation spring combinations, offset shroud CG and separation springs misaligned. Figure 10-4 shows the vector sum of the pitch and yaw effects and indicates the desired clearance band of the shroud aft ring with the spacecraft. Figure 10-5 shows the clearance reduction (in the yaw plane) of the aft ring of the shroud as it traverses longitudinally. Spring misalignment, residual rate, and fish-tailing effects are shown separately, inasmuch as they are the only parameters that contribute significantly to clearance reduction. Figure 10-6 shows the aft ring clearance reduction for the pitch plane.

10.3 RANGER SHROUD

The Ranger shroud assembly is identified by LMSC-1314301. The assembly is 148.90 in. long and extends from LMSC Station 232.50 to 83.60.* It consists of a 7-degree cone section, a 15-degree cone section and a nose dome. The nose dome is attached to the 15-degree cone section at LMSC Station 93.60. The 15-degree cone section extends from LMSC Station 93.60 to 120.52. It is a magnesium shell with a magnesium ring-stiffener located approximately at its midpoint. The 7-degree cone section extends from LMSC Station 120.52 to Station 232.50 and has a base diameter of 65.70 inches. It is also a ring-stiffened shell structure, having six

*Station numbers are referenced to Agena B in this discussion of the Ranger shroud system.

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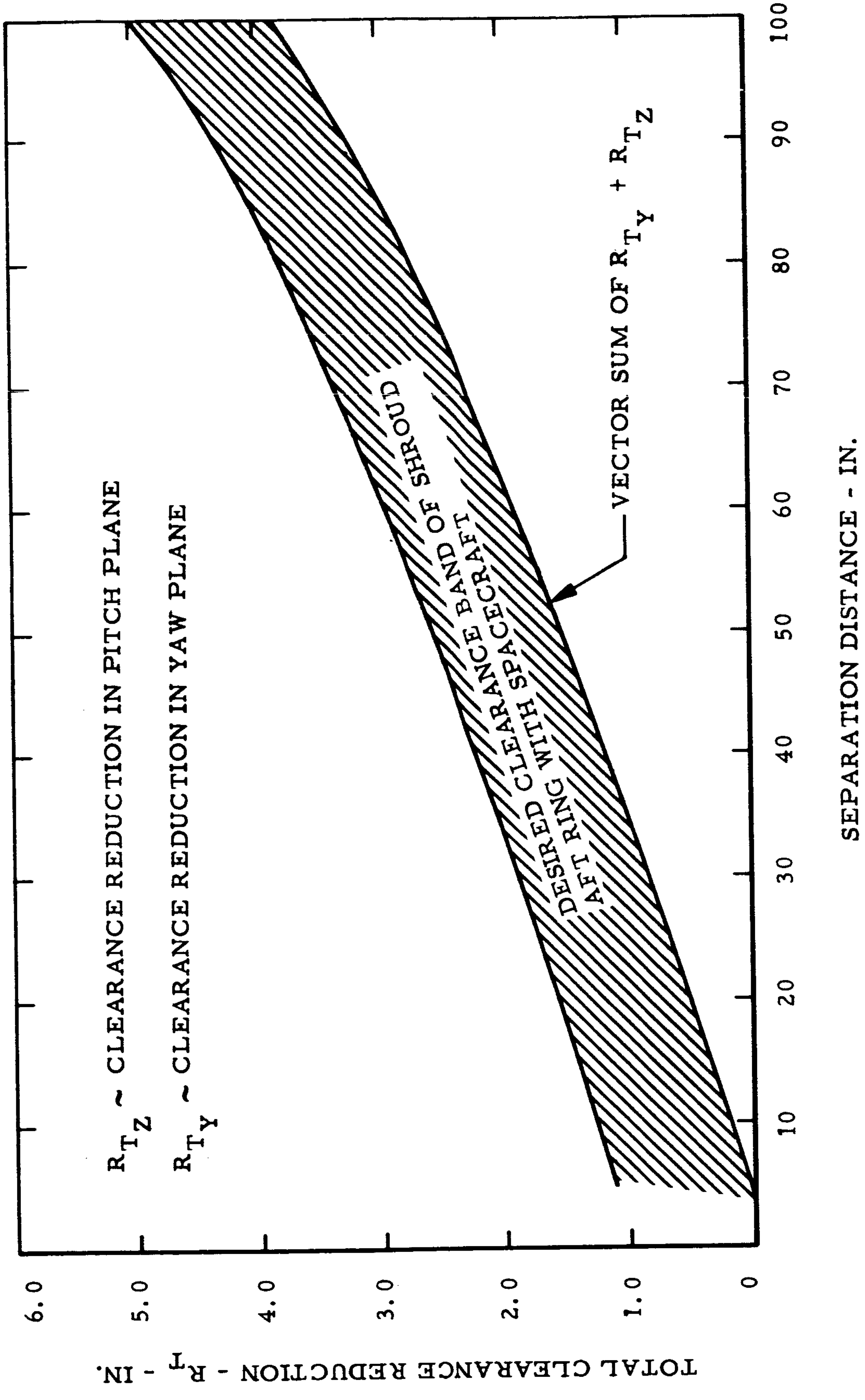

Fig. 10-4 Mariner Mars Shroud Clearance Reduction — Vector Sum of R_{T_Y} + R_{T_Z}

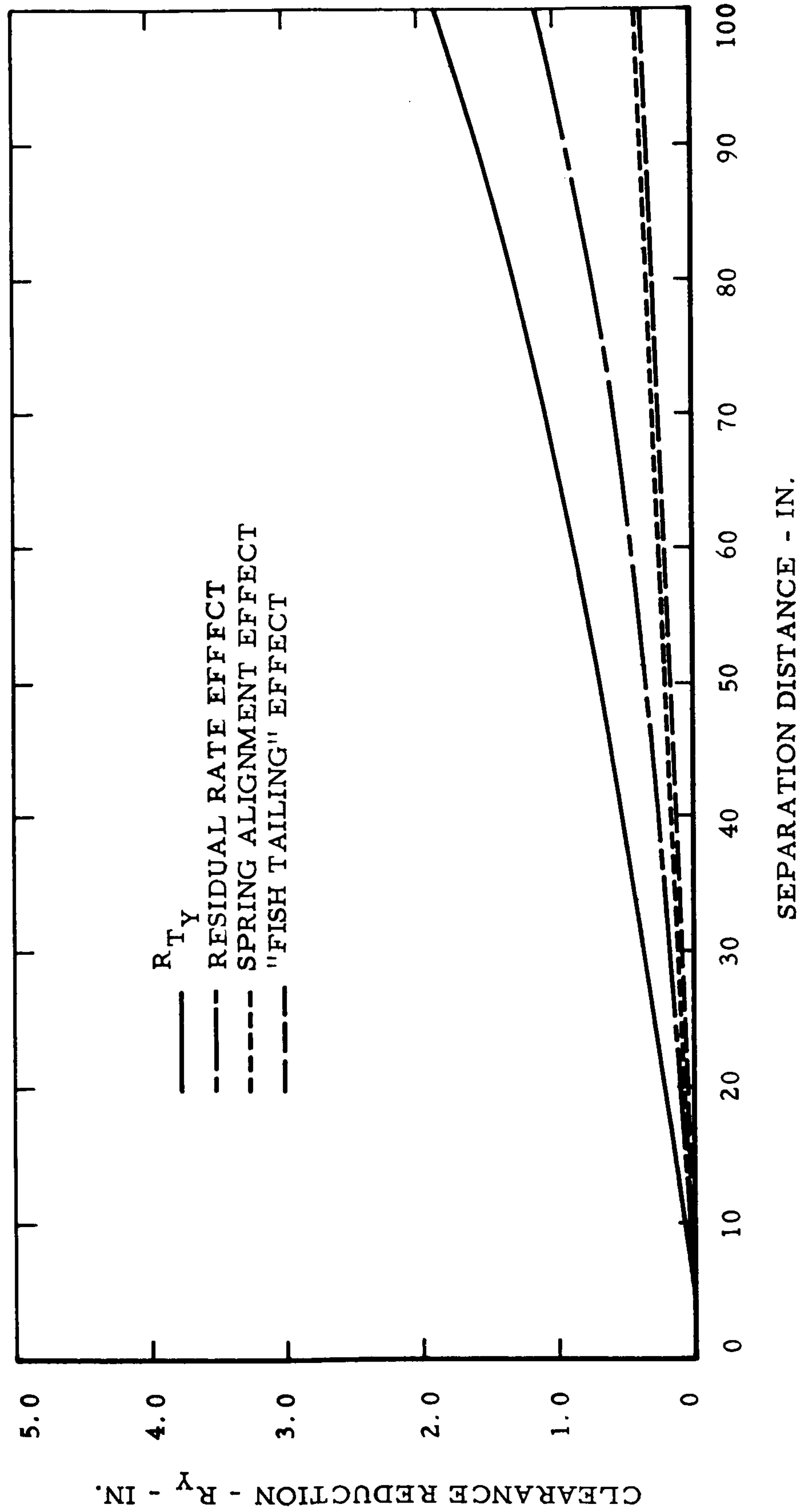

Fig. 10-5 Mariner Mars Shroud Clearance Reduction—Yaw Plane

E-3236-5

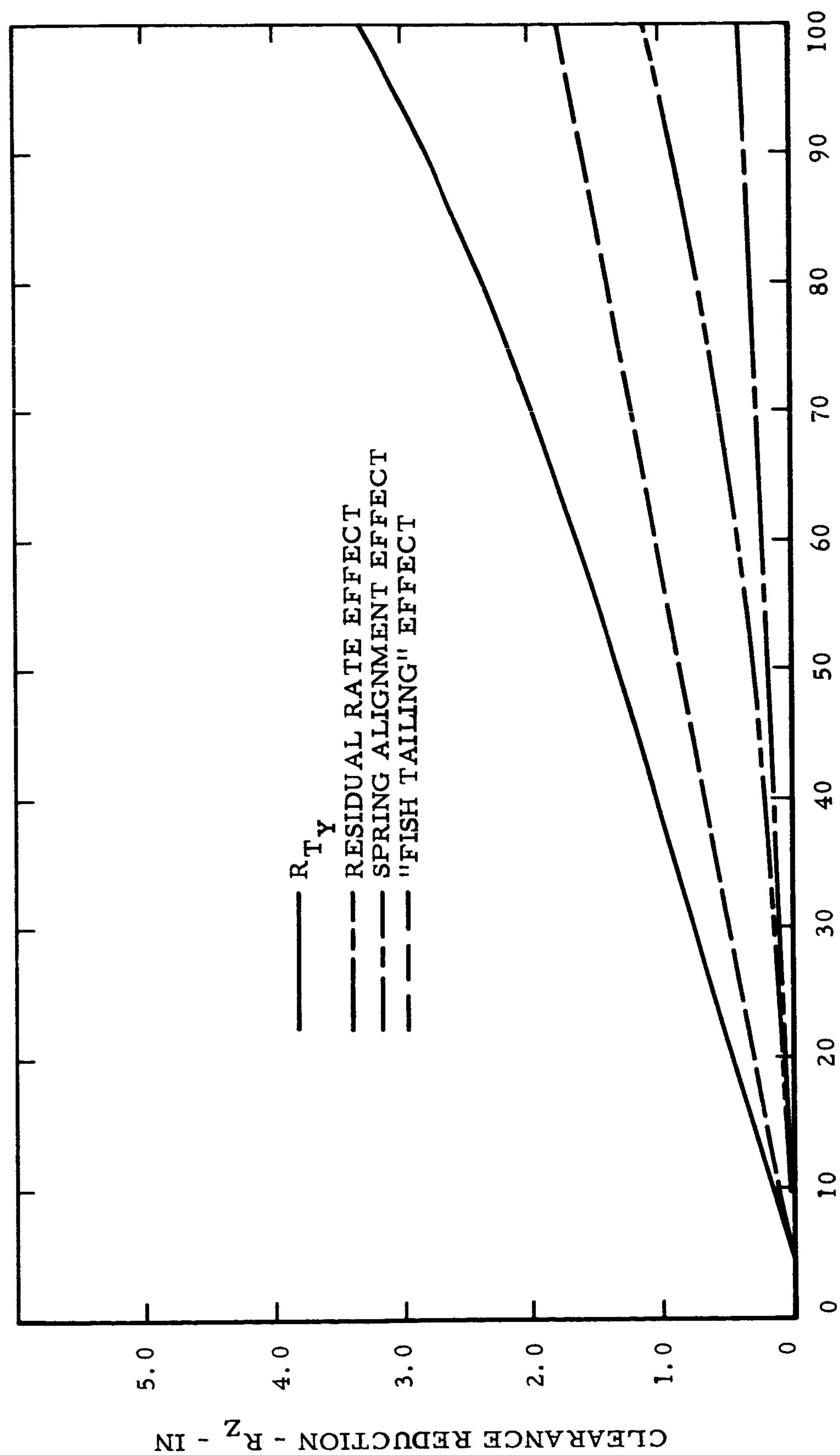

Fig. 10-6 Mariner Mars Shroud Clearance Reduction -- Pitch Plane

intermediate extruded magnesium stiffening rings. The overall shroud dimensions are shown in Fig. 10-7. An aluminum heat shield is provided in the 7-degree cone section to protect the spacecraft and a 0.5-in. thick fiberglass radiation heat shield is provided in the nose cone. The nose dome has a radius of 12 inches and is formed from stainless steel sheet. The shroud is mounted on the adapter at Station 232.50. Pyrotechnically released pin-pullers permit eight springs to eject the nose cone forward over the spacecraft.

Purge fittings are provided in the forward and aft sections of the shroud for nitrogen purging of the shroud cavity. A rubber O-ring provides a mating seal at LMSC Station 232.50. Critical dimensions and tolerances for the combined Ranger shroud and spacecraft adapter are presented in tabular form on Fig. 10-7.

10.4 NIMBUS SHROUD

The Nimbus shroud is identified by LMSC-1461816. The S-27 and OGO programs employ variations of the shroud with the only significant difference being in the thermal heat shielding which is designed to suit the particular mission. The Comsat shroud (LMSC-1461719) is the same basic configuration but is approximately 107 inches shorter and employs one band to hold it together, where the longer version employs two bands. Other minor differences between the Nimbus shroud and the others are as follows:

Comsat

1. Smaller vent holes with different location and no flappers
2. No "Q" door (for air conditioning)
3. No concentricity band* (tension band)

*The aft end of each shroud half has a tension band installed on the external surface. The ends of the bands are riveted to the shroud. At the strap centers an adjusting bolt is tightened to place the bands in tension. During shroud separation, these bands keep the shroud halves in the shape of semicircles, preventing the shroud edges from deflecting inward and possibly damaging the spacecraft.

E-3236-5

S-27

1. No concentricity band (tension band)

The shroud consists of a cylindrical section, approximately 137 inches long (30 inches on Comsat) and 65 inches in diameter; a 15-degree, half-angle cone; and a 12-inch radius nose dome. Two phenolic fiberglass sections are used in conjunction with the fiberglass nose cap. The main structural member is the shroud skin, 0.130-in. thick phenolic fiberglass laminate, stiffened with internal rings. The rings are U-shaped aluminum alloy half-circle segments terminating in aluminum alloy longerons along the parting plane. Shroud/Agena attachment is by means of two half hoops which mate with a ring bolted to the forward midbody. Over-all dimensions and details for the Nimbus and Comsat versions are shown in Figs. 10-8 and 10-9.

The shroud is made up of two halves that join and separate along the longitudinal axis between Stations 31.532 (138.50 for Comsat) and 245.50.* At Station 31.532 (138.5 for Comsat) the joint is along the station plane, normal to the longitudinal axis. The fiberglass dome nose cap is attached to the -Y side of the shroud (Quadrants 3 and 4). Two mechanical latching mechanisms are installed in the cone section at the separation plane to insure against shroud gapping in the section.

Separation is effected in clamshell fashion to eject the shroud from the ascent vehicle. Externally mounted, segmented, circumferential bands joined with explosive bolts restrain the forward end and mid-section until shroud ejection. The aft end of the shroud is held in position on the adapter ring by explosive bolts installed in integral fittings on the two halves. Positive shroud separation is accomplished by means of two spring-loaded actuators mounted forward of the shroud center of gravity. Pivot fittings located at the rear of the shroud (Station 244.50) insure forward end opening

*Station numbers referenced to Agena B.

of the shroud halves. An external concentricity hoop is provided at the aft end of the shroud to maintain shroud shape. The hoop is split so that it does not span the separation joint.

Venting of the shroud cavity during ascent is accomplished by vent holes covered by "flapper valves" to insure one-way flow.

An air conditioner umbilical coupling is provided in the shroud for internal spacecraft cooling. Conditioned air is vented through a spring-loaded door ("Q" door) located at the aft end of the shroud. (Ground cooling performance is discussed in Section 8.)

10.4.1 Critical Dimensions

Critical dimensions and tolerances for the combined Nimbus shroud and spacecraft adapter are presented in tabular form in Fig. 10-8. The same data are provided for the Comsat configuration in Fig. 10-9.

10.4.2 Flight Clearance Envelope

Because of the small dimensional clearance between the Nimbus shroud and payload, it has been necessary to study the radial deflections that may occur during ascent to determine that no impact between the spacecraft and shroud will occur. To be of value, the spacecraft excursions must be considered along with the shroud deflection in establishing the total clearance envelope required. The calculations of the flight envelope for the Nimbus shroud are given in three parts, manufacturing tolerance, vibration dynamics during the boost phase of flight, and air-load bending induced during ascent to approximately 60,000 ft.

Manufacturing tolerances for the Nimbus shroud vary from 0.125 to 0.826 inches and are shown by station in Fig. 10-10. These manufacturing tolerances can also be applied to the S-27 and OGO shrouds and can be used as a guide for the Comsat shroud.

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The vibration excursions of various points on the shroud are calculated values and are presented as radial deflections at the stations listed in Fig. 10-11. These radial excursions will vary with the weight and dynamic characteristics at the spacecraft because of the dynamic interaction of the two systems at the common Agena attachment plane. For this reason, these values may be used as a guide only, exact values must be determined for each payload application involving critical spacecraft/shroud clearances.

Bending loads are calculated values based on flight aerodynamic pressures predicted for the Nimbus ascent profile. The calculated values of the radial deflections due to bending loads are also presented in Fig. 10-11. The bending (static) deflections, like the vibration excursions, are mission-sensitive and will vary with the trajectory and booster system employed.

The total expected radial excursion of the Nimbus shroud, when used on the Nimbus mission, can be obtained by adding the radial manufacturing tolerance, the radial vibration excursion, and the radial static deflection presented in Figs. 10-10 and 10-11. Values are presented for both the X-X and Y-Y axis to account for the structural variations between the separation plane and the perpendicular axis.

10.5 OAO SHROUD

The OAO shroud covers the entire spacecraft plus Agena and is a three-piece design consisting of a 14 1/2-degree cone-cylinder nose-fairing, GDA 27-76202, an aluminum alloy mid-fairing, GDA 27-76104, and a stainless steel aft-fairing, GDA 27-76102. The nose-fairing and mid-fairing are assembled as a unit extending from the forward end of the nose cone to the aft restraining fittings at LMSC Station 361.00* (GD/C Station 383.42). The assembly joint of this unit with the aft fairing is at LMSC Station 247.58 (GD/C Station 270.00). This forward unit is of clamshell opening design. The shroud is equipped with two work platforms and access doors for on-the-pad servicing

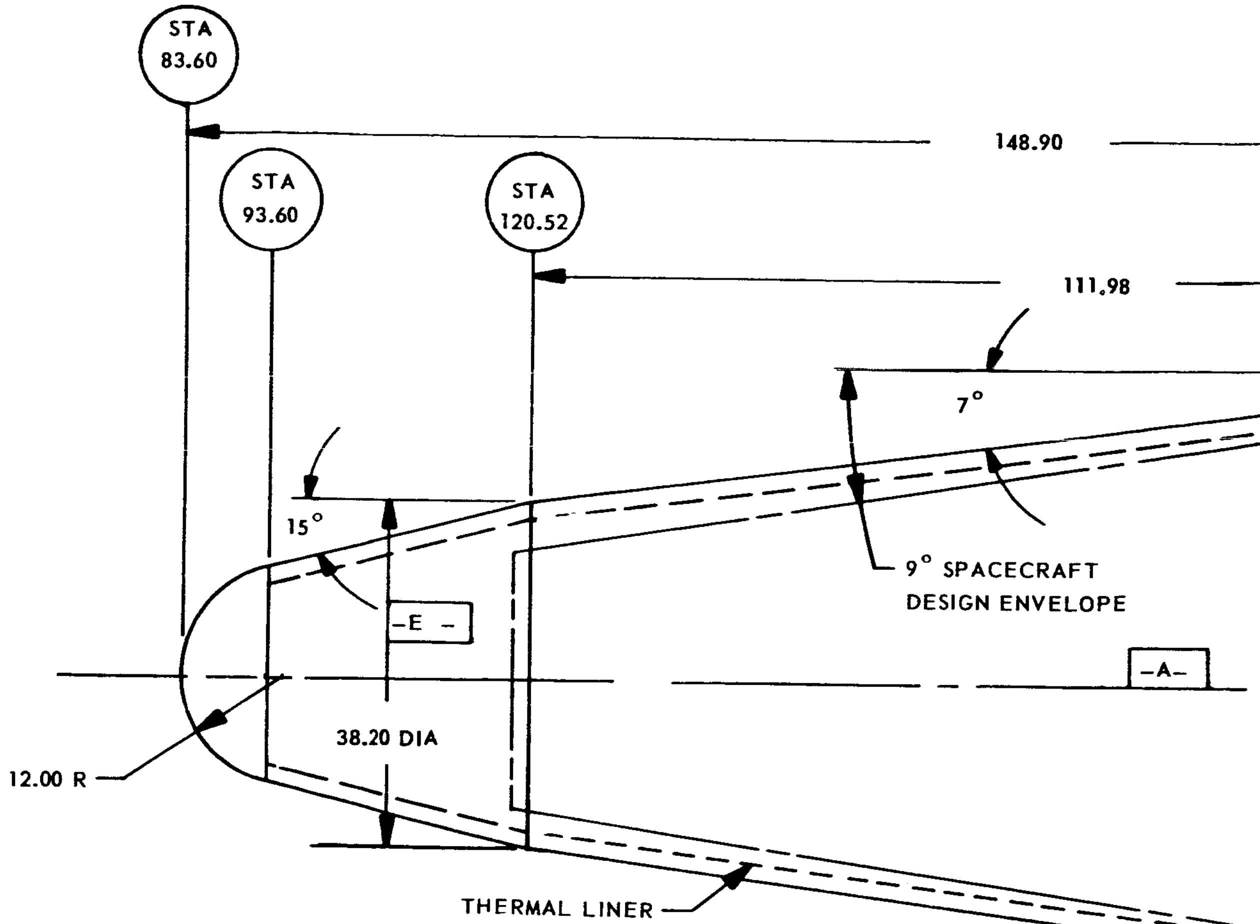
*Referenced to Agena D.

of the Agena and spacecraft, and has fiberglass chutes for installing and withdrawing the Agena umbilicals.

The mid and aft, semi-monocoque, cylindrical fairings provide a constant 10-foot diameter from the Atlas booster to the nose-fairing structure.

Metal sections of the shroud are fabricated from rings, hat-section stringers, and outer skins. They are identified as the aft-and mid-fairings and enclose the Agena D vehicle and Atlas/Agena adapter. The nose-fairing structure consists of a 10-foot cyclindrical section, a conical section and a nose dome; and is RF transparent. The general shroud configuration, with a table of critical tolerances and alignments for the combined shroud and spacecraft adapter, is shown in Fig. 10-12.

The OAO spacecraft is encapsulated in the nose section in the experiment area which is kept free of contamination by a sealing diaphragm and nitrogen purge. The encapsulated spacecraft is transported to the launch pad for mating to the launch vehicle. The aft-fairing is permanently bolted to the Atlas vehicle and remains with the booster following the Agena separation. The nose section is connected to the mid-fairing to make up the clamshell (separable) shroud and is separated by detonation of 22 explosive nut mechanisms along the separation seam, after which spring thrusters mounted adjacent to the dome structure on each shroud half separate the shells. Pivot restraining fittings located at the rear of the shroud assembly (LMSC Station 361.00; GD/A Station 383.42) assure nose-first opening. Figure 10-13 illustrates the clamshell separation of the shroud.



BLOCK LETTER	DESIGNATION	STATION LMSC	TOLERANCES	DIAMETER
H	MOUNTING SURFACE OF AGENA	224.50	⊥ B .020 TIR	60.00
D	MOUNTING SURFACE OF SPACECRAFT ADAPTER	244.50	⊥ A .005 TIR	60.00
C	MOUNTING SURFACE OF SHROUD	232.50	∥ D .005 TIR	65.70
C	MOUNTING SURFACE OF SHROUD	232.50	⊥ A .005 TIR	65.70
G	INSIDE DIA OF SPACECRAFT – AGENA ADAPTER	244.50	⊥ A .010 TIR	56.40
F	INSIDE DIA OF SPACECRAFT – SHROUD ADAPTER	232.50	⊥ A .005 TIR	62.00
D	MOUNTING SURFACE OF SPACECRAFT ADAPTER	244.50	⊙ C .010 TIR	60.00
E	OUTSIDE DIA OF SHROUD	120.52	⊙ C .060 TIR	38.20
E	OUTSIDE DIA OF SHROUD	120.52	∥ C .040 TIR	38.20

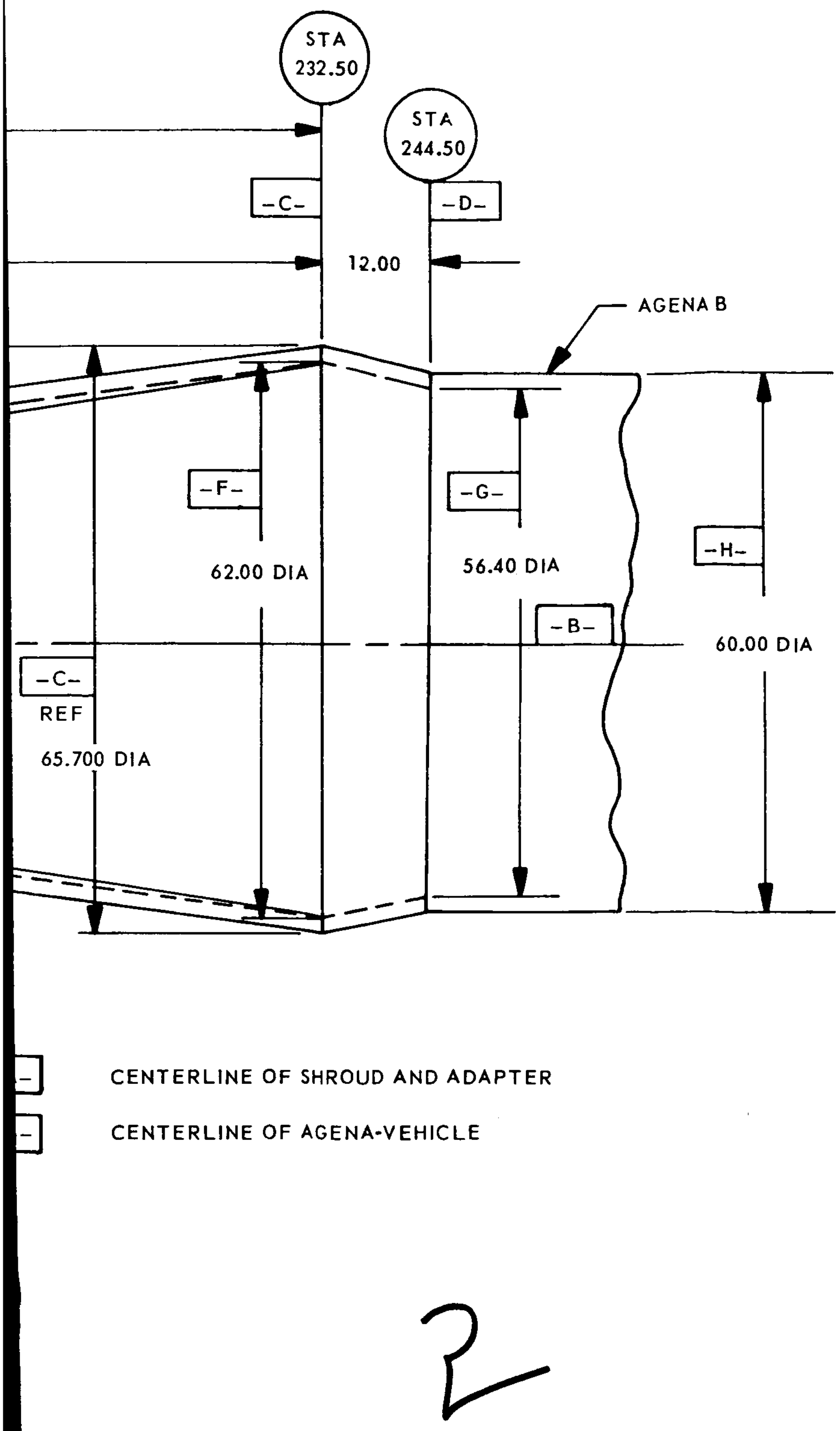
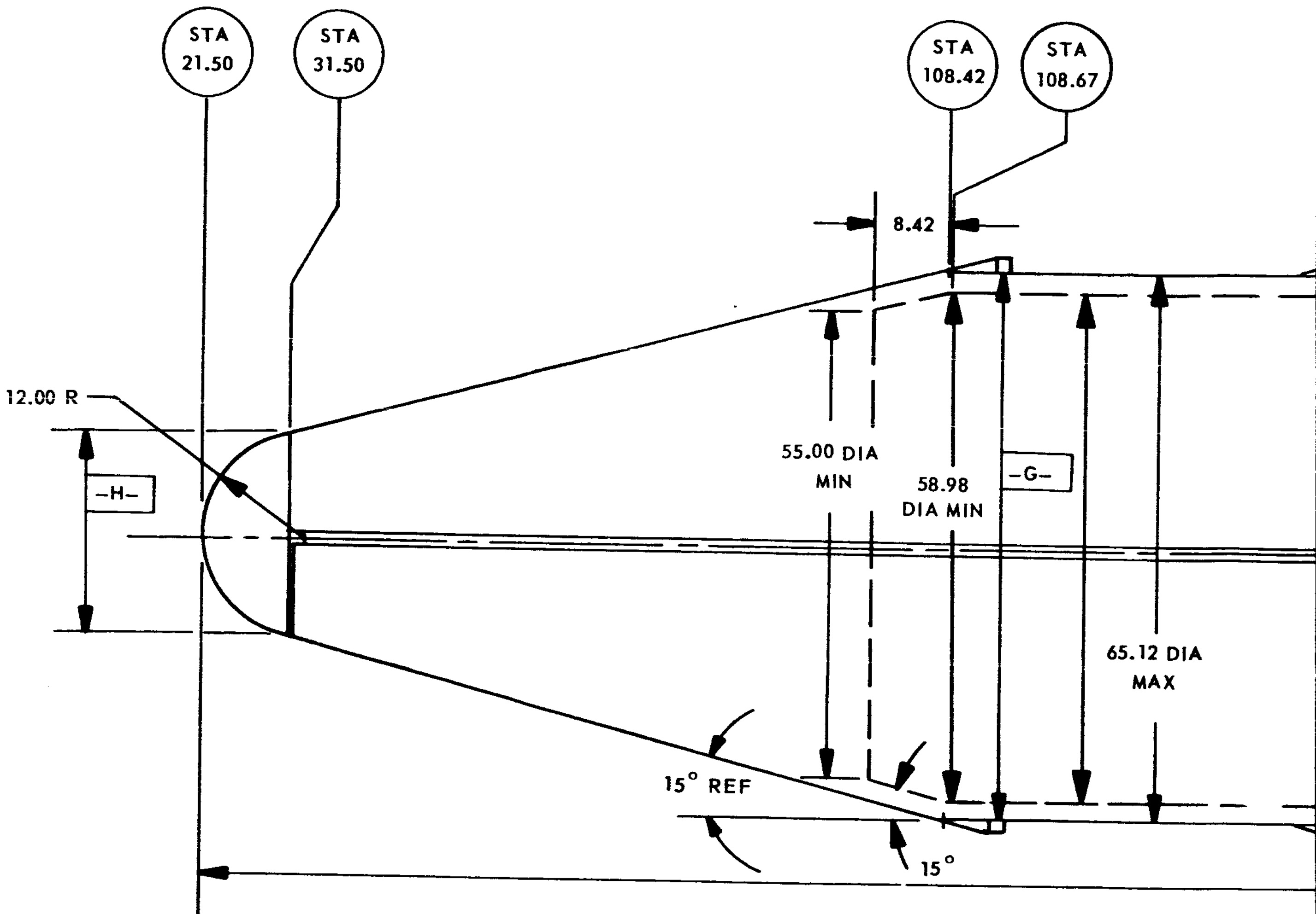


Fig. 10-7 Ranger Shroud System Overall Dimensions and Tolerances

E-3236-5



BLOCK LETTER	DESIGNATION	STATION LMSC	TOLERANCE	DIAMETER
C	MOUNTING SURFACE OF SHROUD	244.50	⊥ A .020 TIR	60.80
J	MOUNTING SURFACE OF ADAPTER	244.50	⊙ C .010 TIR	60.00
E	OUTSIDE DIA OF SHROUD	245.50	⊙ J .180 TIR	65.00
K	INSIDE DIA OF SHROUD	245.50	⊙ C .090 TIR	60.31
F	DIA OF MID STRAP ASSY	157.00	⊙ C .340 TIR	65.00
G	DIA OF FWD STRAP ASSY	115.00	⊙ C .500 TIR	65.00
H	OUTSIDE DIA AT BASE OF NOSE CAP	31.50	⊙ C .250 TIR	—
D	MOUNTING SURFACE OF AGENA	245.50	⊥ B .020 TIR	60.00
F & G	OUTSIDE DIA OF SHROUD	157.0 & 157.0	∥ C .040 TIR	65.00
L	B. C. DIA OF ADAPTER	244.50	⊙ D .010 TIR	58.546

-A- CENTER

-B- CENTER

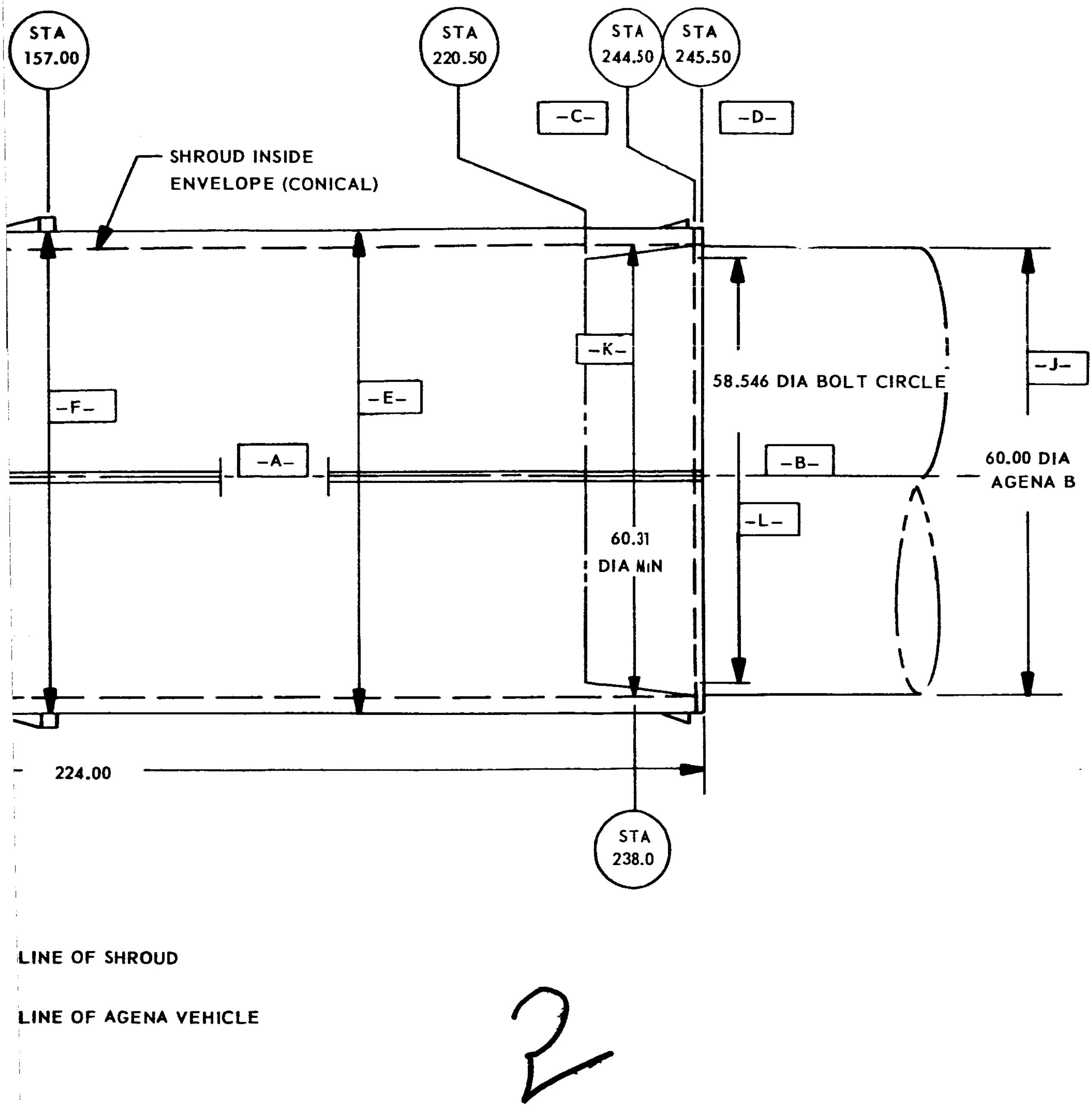
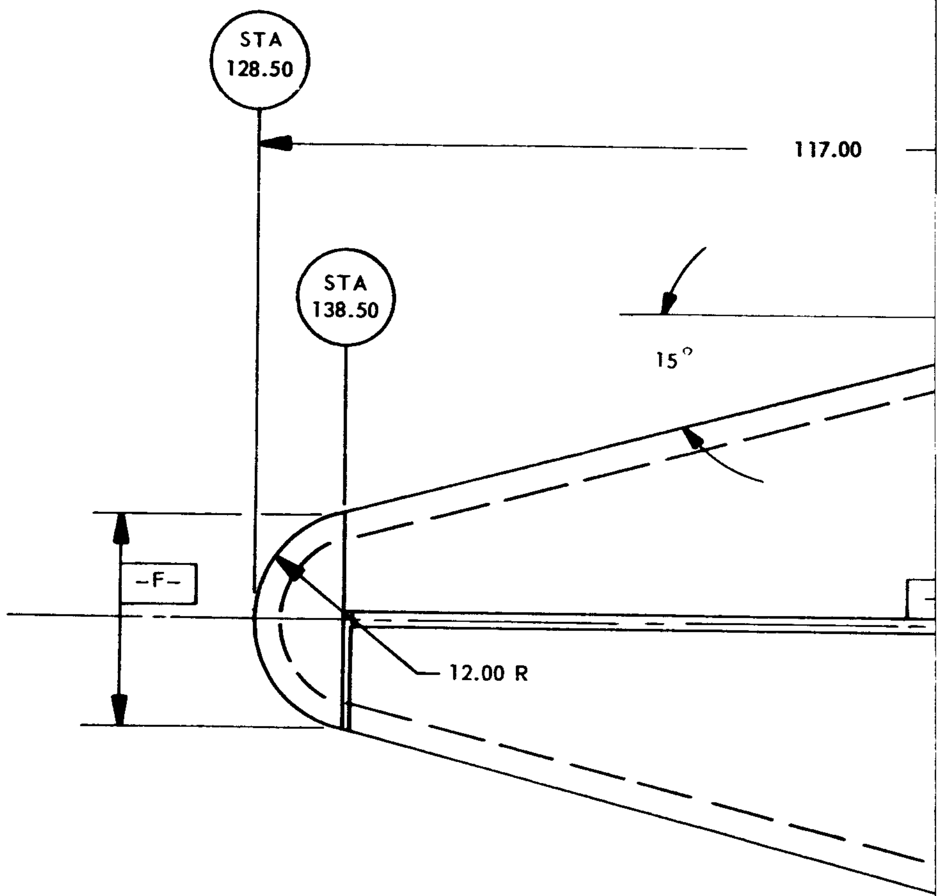


Fig. 10-8 Nimbus Shroud System Overall Dimensions and Tolerances

E-3236-5



BLOCK LETTER	DESIGNATION	STATION LMSC	TOLERANCE	DIAMETER
E	MOUNTING SURFACE OF AGENA	245.50	⊥ B .020 TIR	60.00
C	MOUNTING SURFACE OF ADAPTER	244.50	⊙ E .010 TIR	60.00
D	MOUNTING SURFACE OF SHROUD	245.50	⊥ A .020 TIR	64.80
F	SHROUD DIA AT NOSE CAP	138.50	⊙ C .250 TIR	—
C	MATING SURFACE OF SHROUD	244.50	~ .015 TIR	64.82

-A- CENTERL

-B- CENTERL

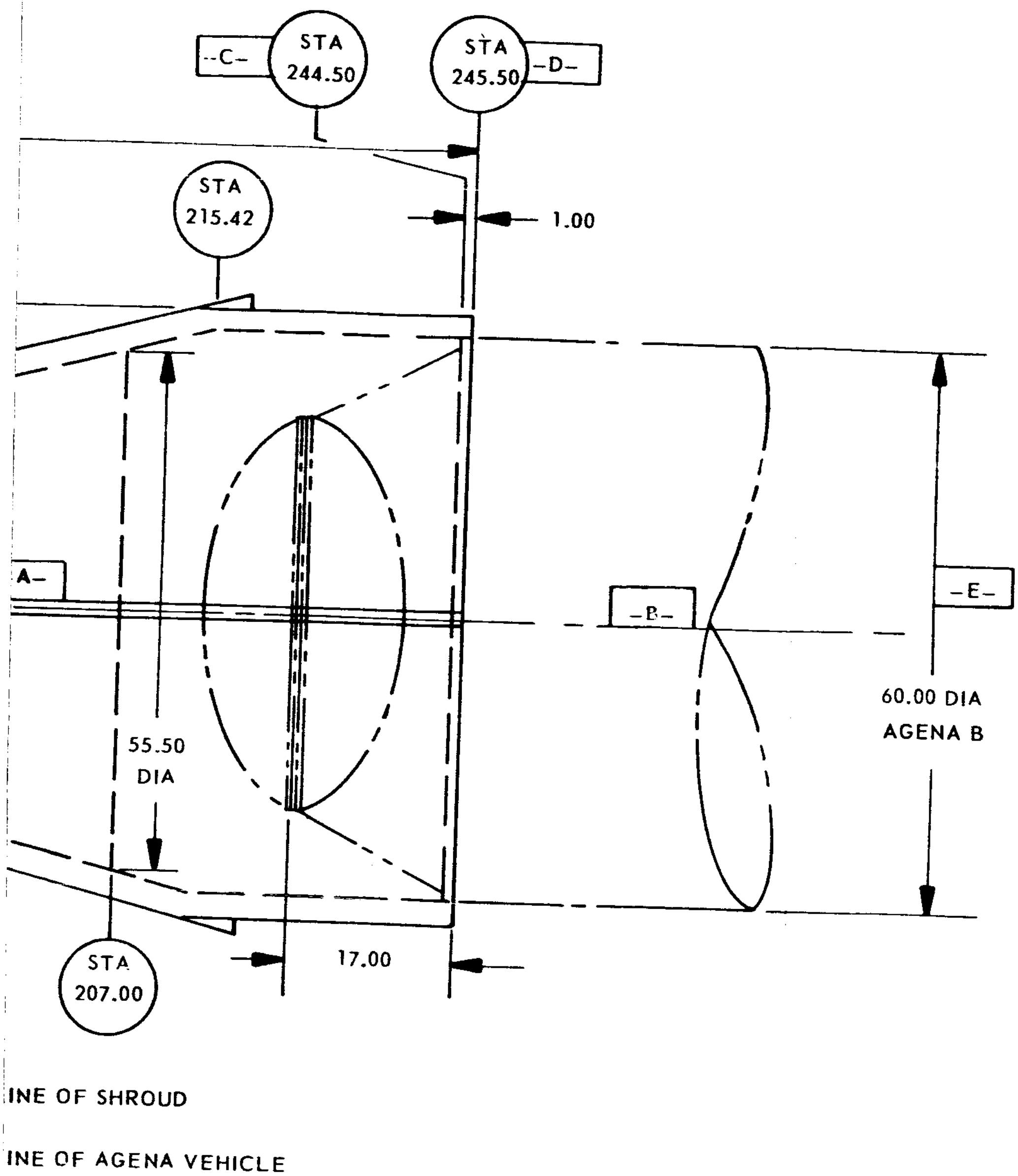


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Fig. 10-9 Comsat Shroud System Overall Dimensions and Tolerances

E-3236-5

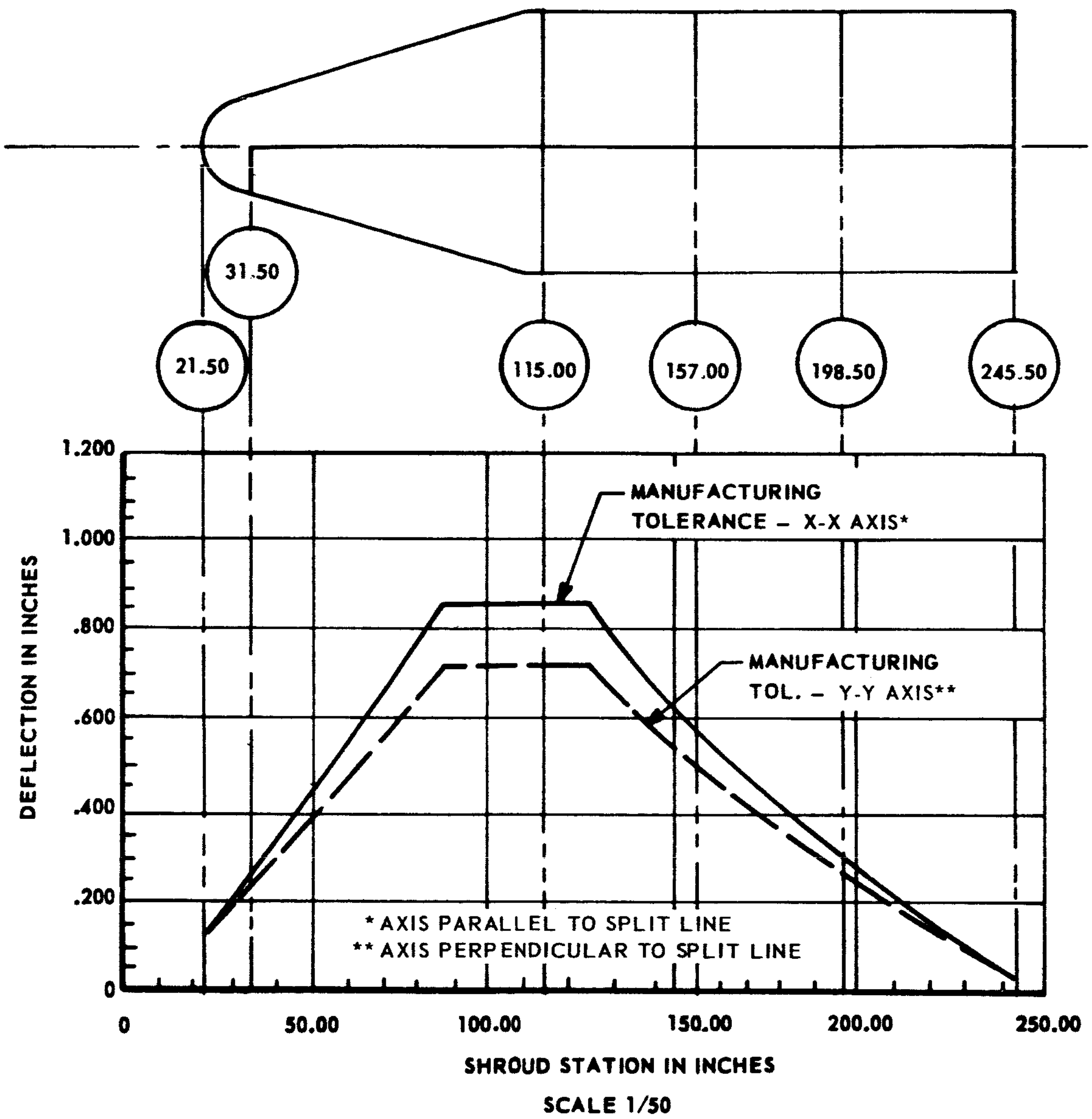


Fig. 10-10 Nimbus-Type Shroud Manufacturing Tolerances (General Application)

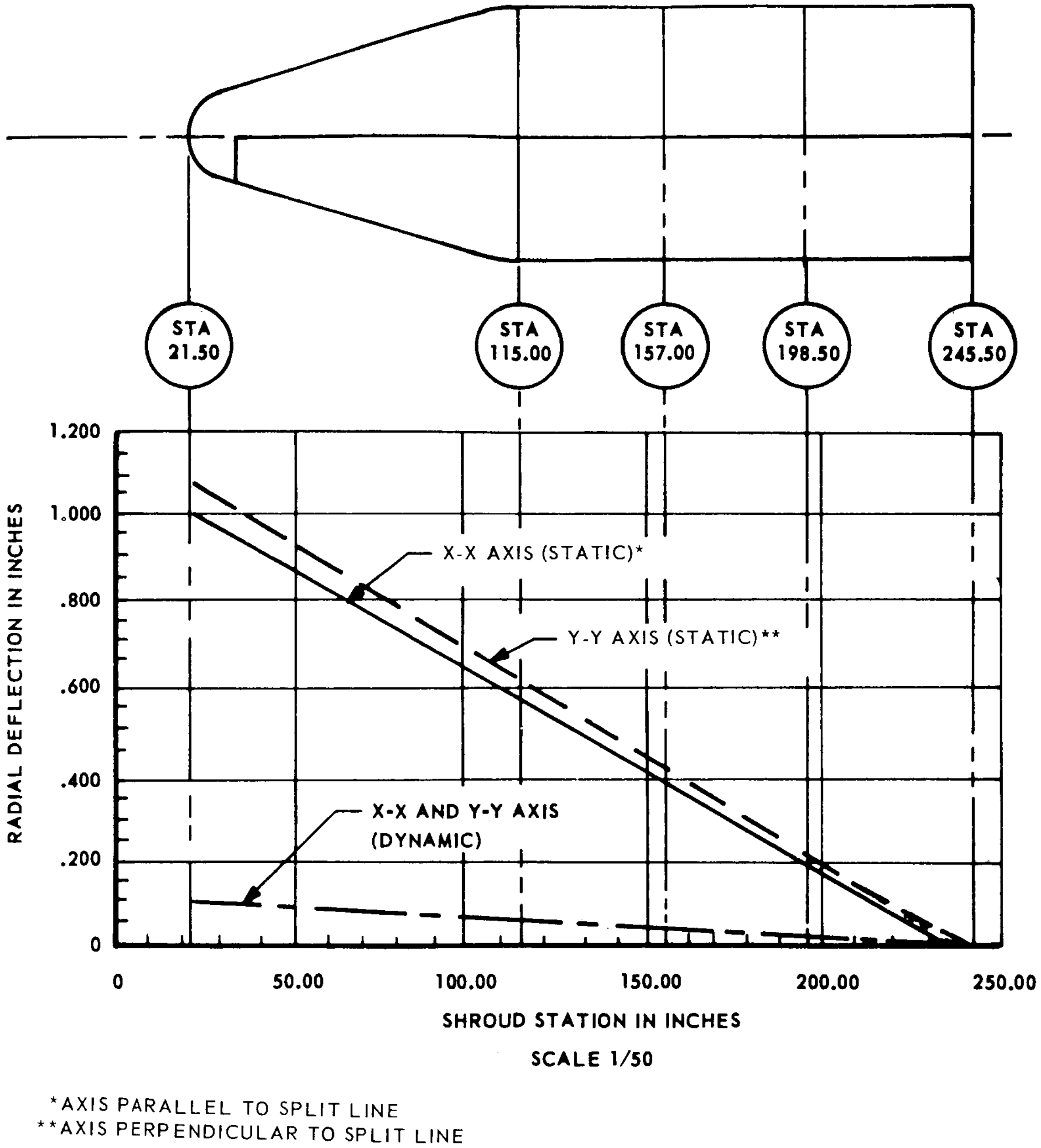
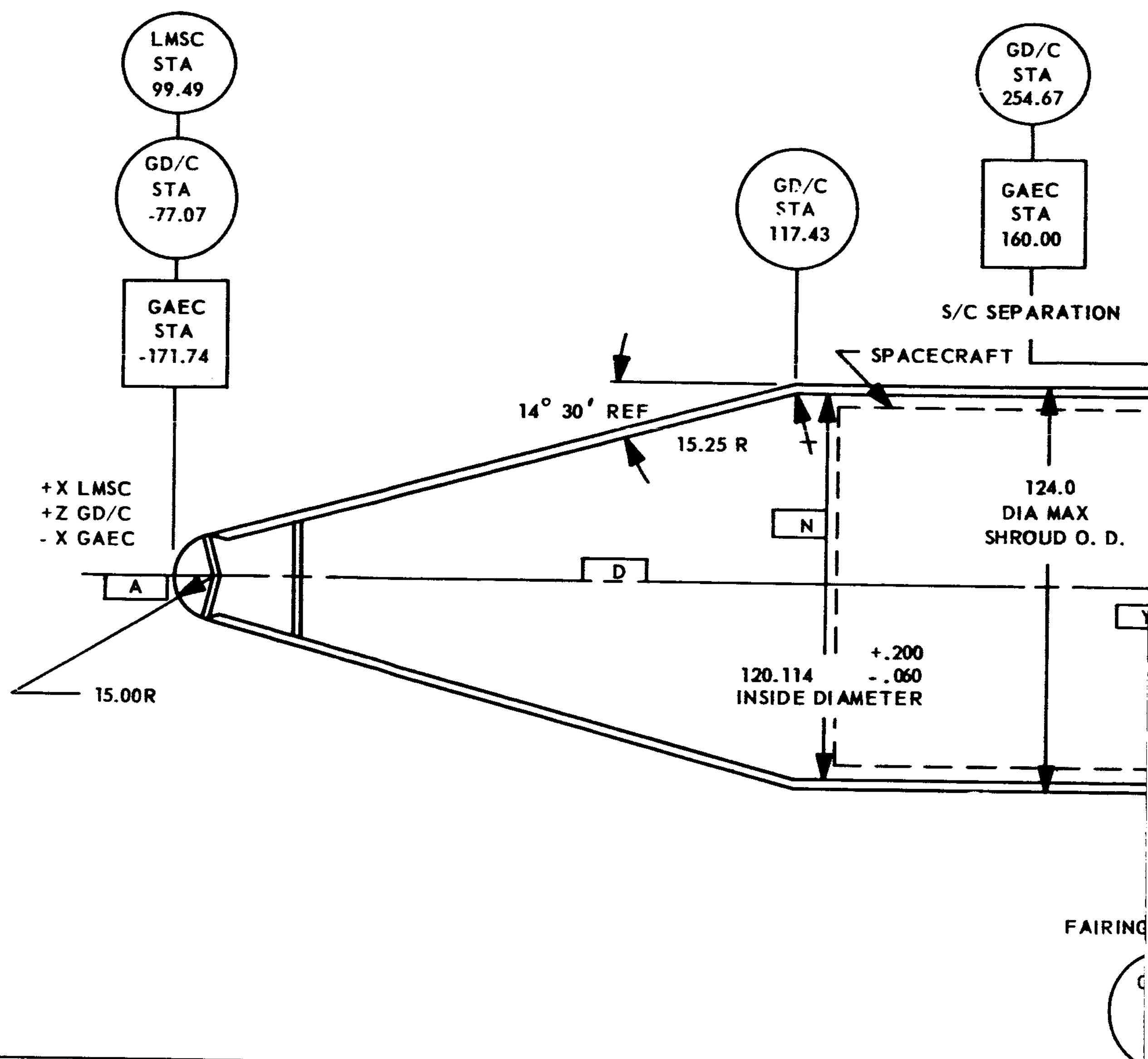


Fig. 10-11 Nimbus Shroud Dynamic Envelope (Nimbus Application Only)



BLOCK LETTER	DESIGNATION	STATION			TOLERANCES AS RELATED TO ASSEMBLED LIMITS	DIAMETER
		GD/C	LMSC	GAEC		
U	BOLT CIRCLE OF NOSE CONE	270.00			⊙ D .010 T.I.R.	– –
V	BOLT CIRCLE OF AGENA		246.35		⊙ Y .035 T.I.R.	58.00
X	MOUNTING SURFACE OF SPACECRAFT ADAPTER			174.10	⊥ D .025 AT BOLT CIRCLE Y	60.00
Y	BOLT CIRCLE OF SPACECRAFT ADAPTER			174.10	⊙ D .025 T.I.R.	58.00
N	INSIDE DIAMETER OF NOSE CONE	125.67			⊙ U .020 T.I.R.	120.114

GD/C
LMSC
GAEC

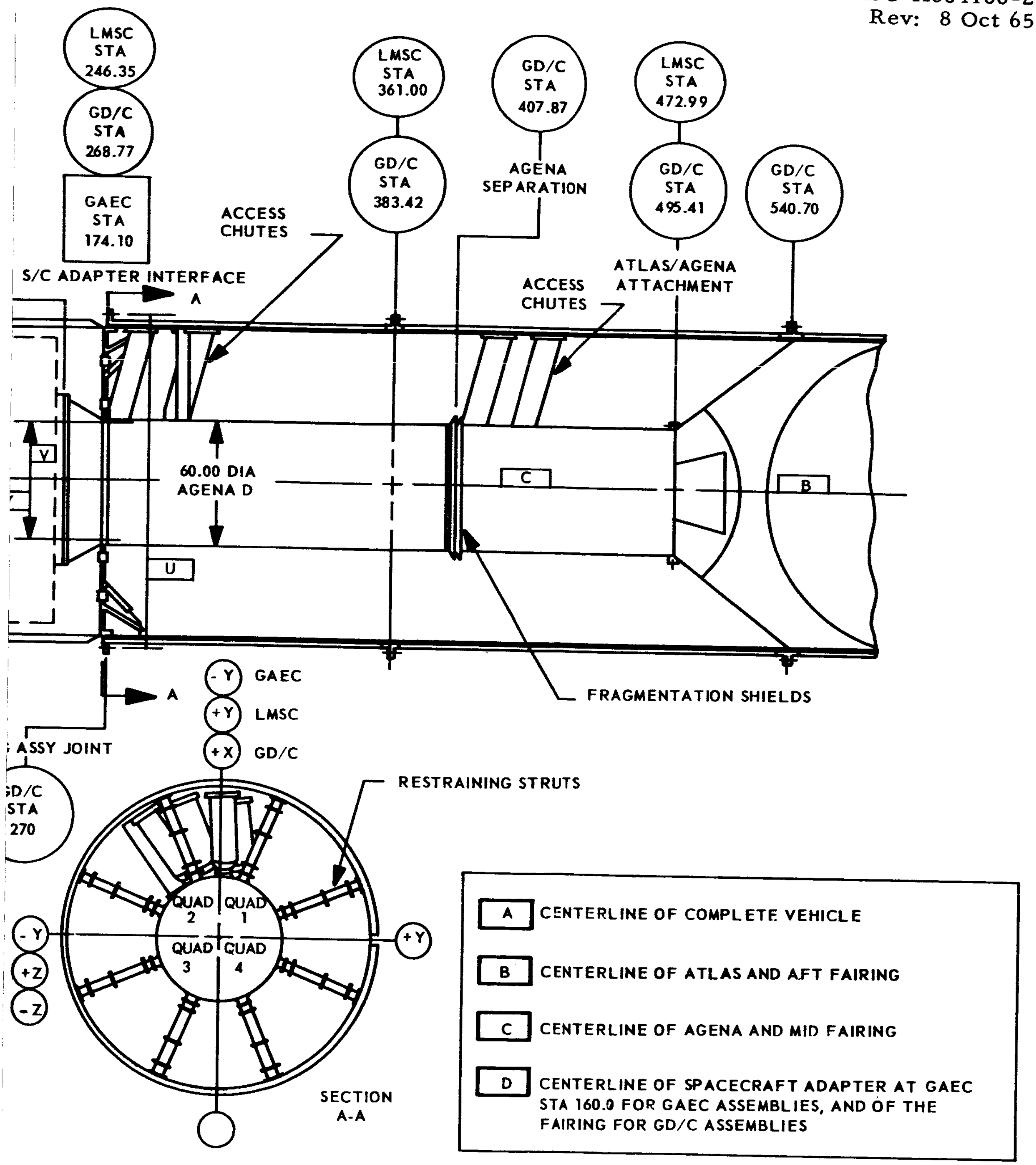


Fig. 10-12 OAO Shroud System Overall Dimensions and Clearances

2

E-3236-5

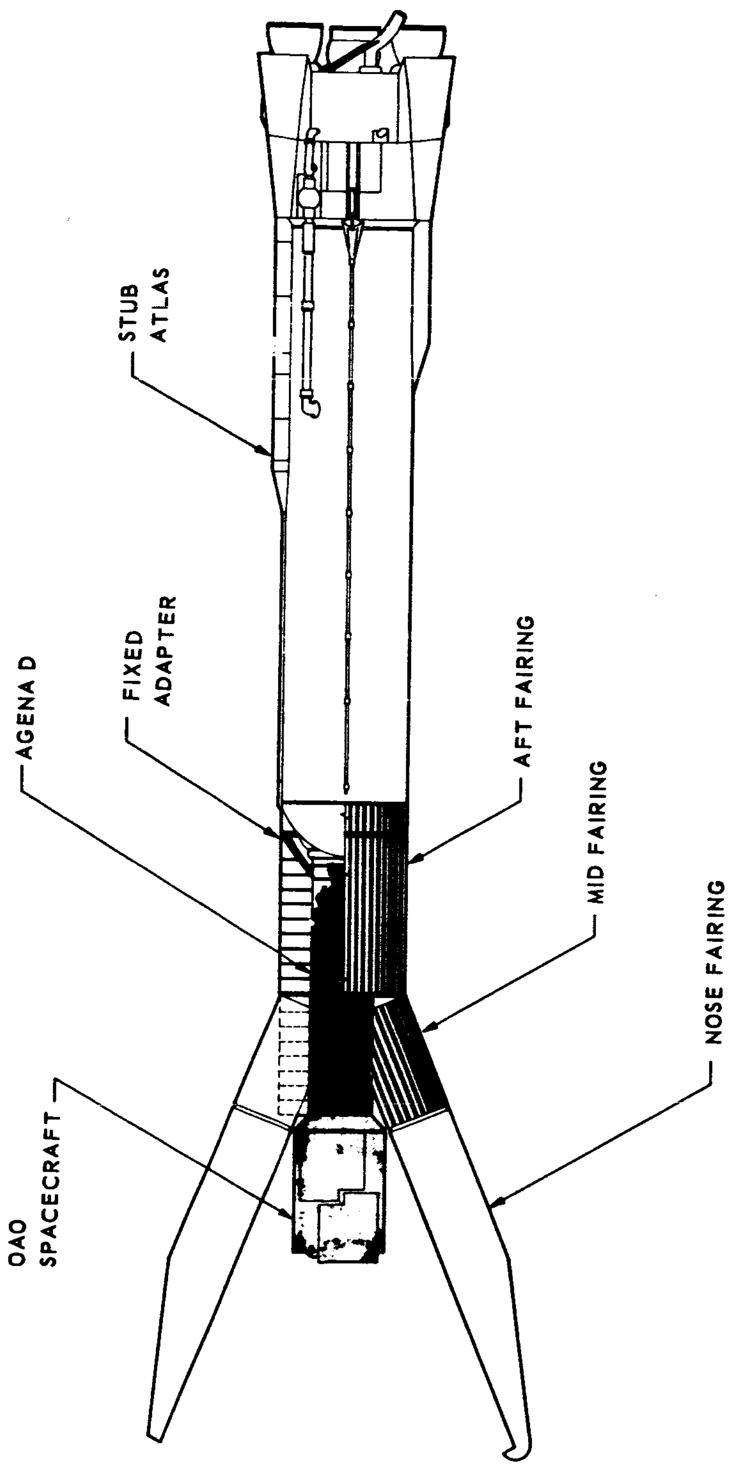


Fig. 10-13 OAO Shroud Separation (Clamshell Section)

10.6 STANDARD AGENA CLAMSHELL SHROUD

The Standard Agena Clamshell Shroud is an improvement of the Agena Long Shroud (Nimbus and OGO type). The shroud design incorporates a new shroud mounting and hinge system, a new separation spring system, and a new nose latch system. These improvements provide a shroud that is easier to install and that permits mating of the spacecraft/adapter/shroud to the Agena as an assembled package. The shroud system includes a shroud/spacecraft/Agena adapter ring (called shroud transition ring) that attaches to the Agena by an external bolt flange similar to the Mariner design and has provisions for attachment and structural support of the spacecraft and shroud. The shroud transition ring contains a diaphragm that provides for complete encapsulation of the spacecraft, with the shroud in place, prior to mating to the Agena. The structure of the shroud—fiberglass shell, internal metal frames and longerons—is similar to the POGO shroud. The height of the shroud above the Agena interface has been increased by two inches beyond that of the POGO shroud. This added length results in the same shroud height above the top of the shroud transition ring as previously existed above the top of the Agena for the Nimbus shroud; i.e., the transition ring is two inches high.

The top flange of this transition ring contains 16 bolt holes for attachment of a spacecraft adapter. The shroud attaches to the exterior flange portion of the transition ring by means of a V-band type clamp. Eight separation springs contained in the base of the shroud, four in each shroud half, act against the V-band flange on the transition ring. Hinges are incorporated at the base of the shroud halves. Hinge loads are absorbed by fittings attached to the transition ring. The hinge is designed so that the shroud halves are held captive for the first portion of rotation, after which time the shroud halves are free to rotate off the hinge.

E-3236-5

The two flat bands of the POGO shroud have been retained. The mechanical nose latch used at the top of the POGO shroud has been replaced by a more reliable system. This new system is a simple tie bar between the two shroud halves. The tie bar is cut at separation by cable cutters — two cutters have been incorporated for redundancy. Shroud separation is accomplished by pyro-release of the V-band, the two flat bands and the cutting of the nose tie bar. The springs can then impart rotation to the shroud halves and separation is achieved.

The basic configuration and dimensions for the shroud are shown in Fig. 10-14. The shroud total weight including all separated and non-separated items is estimated to be 643 lb plus thermal insulation weight. (This weight includes all items shown in Fig. 10-14 forward of Station 247.0). Drop weight is estimated to be 597 lb (shroud halves, V-band and flat bands) plus thermal insulation. Thermal insulation weights are not included in these two weight estimates because insulation weight will depend on the requirements of each mission. (Weight of the insulation on the POGO shroud is approximately 52 lb). More definitive shroud weights will be available after fabrication and qualification of the new shrouds.

A preliminary shroud internal envelope for Atlas/Agena boosted vehicles and for TAT/Agena boosted vehicles are tabulated in Tables 10-1 and 10-2 respectively. The reader is cautioned that these dimensions were determined from design drawings prior to fabrication or qualification testing of these shrouds. These dimensions must not be used for any purpose other than very preliminary planning, and it is mandatory that LeRC be contacted before any further use is made of these dimensions. Accurate values for these envelopes are expected to be available from shroud qualification tests scheduled for approximately the first quarter of calendar year 1966.

PRELIMINARY

Table 10-1

ENVELOPE FOR ATLAS/AGENA BOOSTED VEHICLES

Standard Agena Clamshell Shroud Internal Envelope

LMSC Station (In.)	Static[1] Dimension Dia (In.)	Static Load[2] Deflection Radial (In.)	Dynamic Load[3] Deflection Radial (In.)	Dynamic Envelope Dia (In.)	LeRC Recommended Envelope Dia (In.)
56.875	34.076	.91	.122	32.012	30.50
108.545	59.990	.66	.084	58.502	57.50
115.125	60.026	.63	.081	58.604	57.61
159.125	60.276	.42	.06	59.316	58.38
195.125	60.376	.26	.04	59.776	59.00
223.625	60.500	.12	.02	60.220	59.50
245.00	60.020	---	---	60.020	60.00

NOTES:
(1) Minimum static dimension at rings after all dimensional tolerances are subtracted.

(2) Static load deflections are based on Nimbus shroud deflection rates and the modified shroud design limit loads for Atlas booster vehicles.

(3) Dynamic load deflections are for a 9-meter per sec (1-cos) gust.

CAUTION: Consultation with LeRC is necessary before commitment for use of envelope because of preliminary nature of information.

PRELIMINARY

PRELIMINARY

Table 10-2

ENVELOPE FOR TAT/AGENA BOOSTED VEHICLES

Standard Agena Clamshell Shroud Internal Envelope

LMSC Station (In.)	Static(1) Dimension Dia (In.)	Static Load(2) Deflection Radial (In.)	Dynamic Load(3) Deflection Radial (In.)	Dynamic Envelope Dia (In.)	LeRC Recommended Envelope Dia (In.)
56.875	34.076	1.17	.122	31.492	30.00
108.545	59.990	.85	.084	58.122	57.00
115.125	60.026	.81	.081	58.244	57.14
159.125	60.276	.54	.06	59.076	58.10
195.125	60.376	.35	.04	59.596	58.90
223.625	60.500	.14	.02	60.180	59.50
245.00	60.020	---	---	60.020	60.00

NOTES:

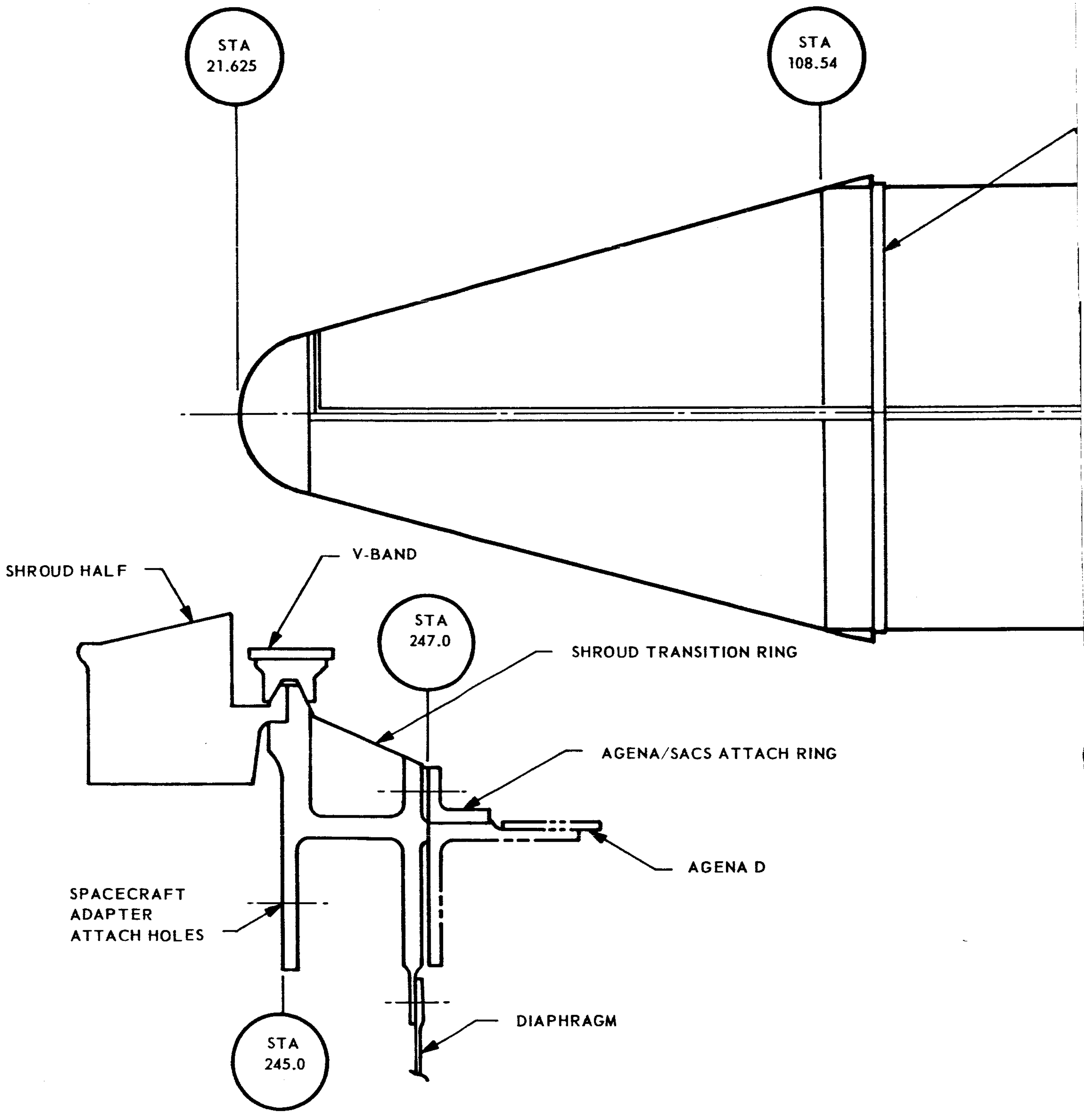
(1) Minimum static dimension at rings after all dimensional tolerances are subtracted.

(2) Static load deflections are based on Nimbus Shroud deflection rates and the modified shroud design limit loads for Atlas boosted vehicles.

(3) Dynamic load deflections are for a 9-meter per sec (1-cos) gust.

CAUTION: Consultation with LeRC is necessary before commitment for use of envelope because of preliminary nature of information.

PRELIMINARY


STA
21.625
STA
108.54
SHROUD HALF
V-BAND
STA
247.0
SHROUD TRANSITION RING
AGENA/SACS ATTACH RING
AGENA D
SPACECRAFT
ADAPTER
ATTACH HOLES
DIAPHRAGM
STA
245.0

SECTION A-A

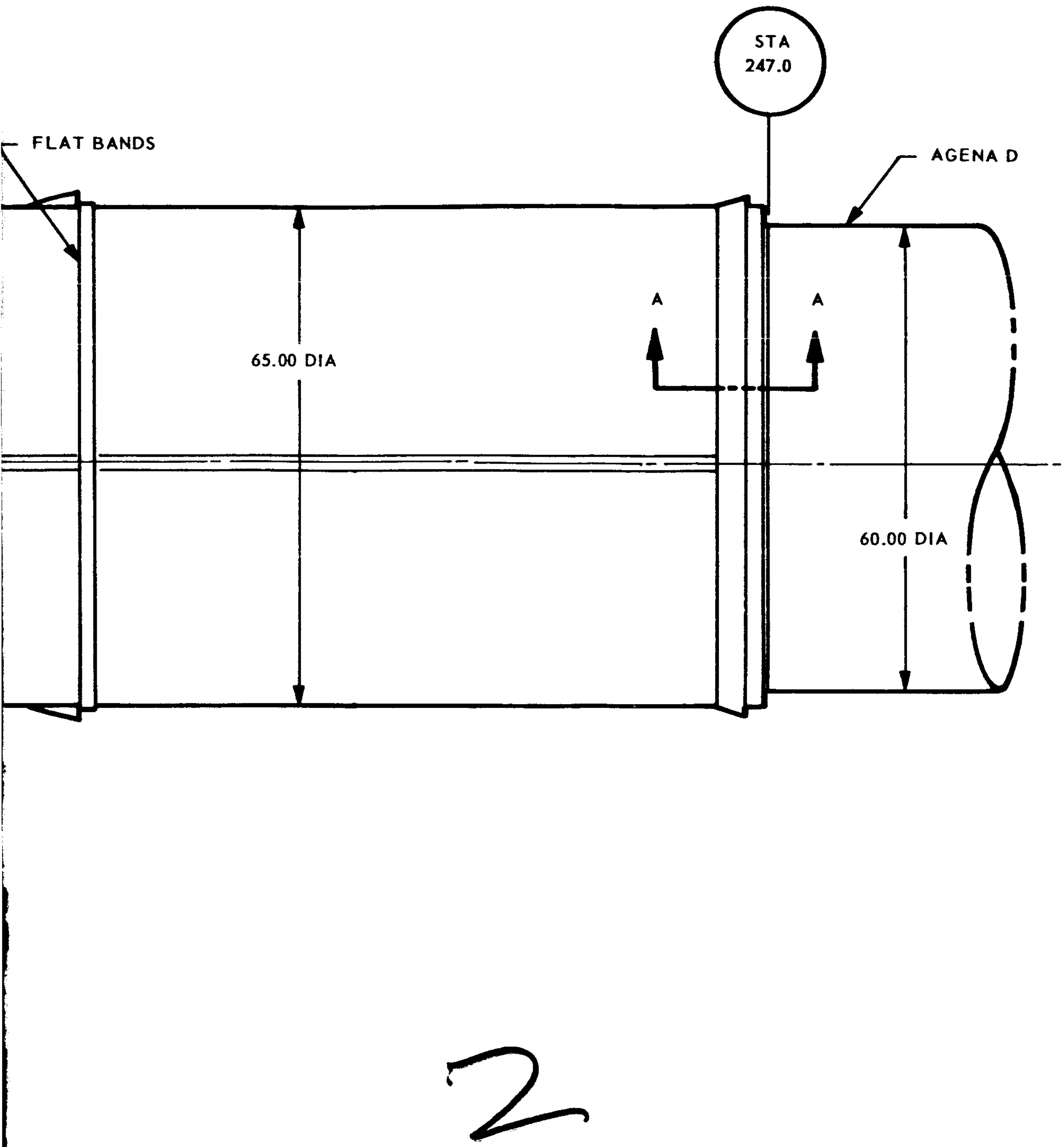


Fig. 10-14 Standard Agena Clamshell Shroud (SACS)

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SECTION 11
SPACECRAFT ADAPTERS

11.1 GENERAL

Every spacecraft employs an adapter system that provides structural support and attachment to the Agena forward rack. On missions employing separable spacecraft*, the adapter may include provision for attachment and separation of the spacecraft and, in some instances, provide interface hardware that initiate or facilitate specific spacecraft events (i.e., spin tables for stabilization spin-up, multiple deployment mechanisms, etc.)

Spacecraft adapters summarized in this section have been employed on NASA Agena missions for separable spacecraft. These systems are those used for the Mariner Mars, Ranger, Nimbus, Echo II, OGO, OAO, and S-27.

11.2 MARINER MARS SPACECRAFT ADAPTER

The Mariner Mars spacecraft adapter is a ring-stiffened shell structure in the form of a short (8.4-in. high) truncated cone (Fig. 11-1). It is made up of a magnesium skin, four magnesium stiffening ribs, four spring brackets, which support the spacecraft separation spring mechanisms, and an upper and lower ring. The lower ring is bolted to the Agena forward rack and provides for the attachment of the magnesium sealing diaphragm and the shroud transition section. The upper adapter ring is made from a magnesium angle and is attached to the magnesium cone. Eight pedestal-type fittings mounted on the upper edges of the adapter provide the interface for mating the spacecraft. The spacecraft mating surfaces of the eight fittings are machined after they are assembled to the adapter to a flatness tolerance of 0.005 inches.

*For some missions the spacecraft or payload remains attached to the Agena which functions as a space platform and orbital vehicle.

The sealing diaphragm attached to the lower adapter ring, is also attached to the Agena forward rack hard points (described in Section 13) by means of four pedestals and a constricting band. Ascent venting of the shroud is through a one-way flapper valve installed in the diaphragm, and opening into the Agena forward rack. Adjustable pads on the spacecraft adapter near the interface provide bearing surfaces for actuators that initiate spacecraft timer start and pyrotechnic arming at separation. Access for servicing the spacecraft ejection springs is provided by four ports in the cone wall. The Mariner Mars adapter structure weighs approximately 57 lb including the spacecraft separation springs, V-band clamp and shroud transition section.

Payload weight-carrying capability of the adapter structure is in excess of 600 lb for the Mariner Mars ascent trajectory. The adapter can be modified to accommodate heavier spacecraft by increasing the material thicknesses of the cone, stiffeners, and rings.

Variations of height and spacecraft mating ring diameter can be accomplished by changing the slope angle of the conical shell. In addition, the spacecraft support pedestals may be changed in size and quantity up to a continuous circular mating ring.

11.2.1 Spacecraft Attachment and Separation Hardware

The eight fittings on the adapter upper ring mate with eight similar fittings located on the corners of the octagonal shaped spacecraft base. The mated assemblies are then secured by the installation of the spacecraft V-band clamp. The spacecraft can be shimmed between the two halves of the clamp fittings to assure true vertical alignment. Torsion stops, which are adjusted during assembly to achieve accurate radial orientation, are provided on the adapter side of each of the 8 pedestals to provide a positive load path for torsional flight loads. These torsion stops are locked after the spacecraft is aligned and before the V-band is brought to full tension.

The V-band clamp system consists of eight V-block attachment fittings made from ZK60A-T5 magnesium, eight bands made from 7075-T6 clad aluminum alloy, six turnbuckles made from 4130 steel (170,000 min. heat treat), ten lanyards (braided nylon cord), and two release assemblies. The release assembly consists of a separation mechanism and two squib port plugs. The separation mechanism consists of a diaphragm, piston, shear washer, retaining nuts, and release bolt. The unit is sealed to prevent contamination of the spacecraft. At spacecraft separation the V-band is released by detonation of a pyro-actuated release assembly which disconnects the band at two places making equal halves. Due to the preload effects of the V-band, the resulting stored energy released causes the band to fly radially outward, thus releasing the spacecraft from the Agena. Lanyards and springs attached to the V-band and anchored to the spacecraft adapter prevent the clamp assembly from striking the spacecraft. Two additional lanyards are attached to the release assemblies to confine the whipping action of the separated band assemblies. The release assembly used on the Mariner Mars Program is also used with other V-band clamp designs on various programs including EGO, POGO, and 823. Figure 11-2 illustrates arrangement of the various components of the V-band assembly.

The spacecraft ejection system consists of 4 spring mechanisms mounted on the spacecraft adapter at 90-degree intervals and approximately 25.5 inches radially from the longitudinal centerline. Each mechanism contains a low alloy steel helical compression spring, a spring shaft, retainer, and linear ball bearing for minimizing lateral friction. Each mechanism also contains adjustment screws for spring stroke adjustment, spring precompression and a planar adjustment for the spring shaft tips.

11.2.2 Mariner Mars Spacecraft Separation

In the Mariner Mars spacecraft separation sequence, detonation of the pyro-operated spacecraft V-band and release assembly allows the spacecraft separation springs to impart a nominal relative velocity to the spacecraft and Agena of 33.2 inches per sec with a maximum tip-off rate of 1.5 degree/sec.

Each spring and its associated mechanism has tolerance values on the spring rates. In addition, the spacecraft and Agena centers of gravity will not lie precisely on the vehicle longitudinal centerline. Thus, due to spring mechanism tolerances, spring tolerances, cg offsets, Agena rotational rate at the time of spacecraft separation, and miscellaneous impulses, a certain misalignment of the resultant separation velocity vector, and a spacecraft rotational rate can be expected. By careful design, installation, and if necessary, balance operations, the velocity vector error and rotational rates can be minimized. The calculated performance of the spacecraft separation system is summarized in Table 11-1. It is based upon approximated Agena and spacecraft inertial characteristics at the separation plane, LMSC Station 238.6, as shown in Table 11-2.

11.3 RANGER SPACECRAFT ADAPTER

The Ranger spacecraft adapter is a fabricated conical assembly that extends from Station 244.50* to Station 232.50 and provides transition from the forward ring of the Agena to the spacecraft interface. It is comprised of two machined, magnesium rings connected by longitudinal magnesium ribs to which removable doors are attached. The outside diameters of the upper ring and lower ring are 65.7 inches and 60 inches, respectively. The 0.071-inch thick removable magnesium doors are fastened to the rings and ribs by means of countersunk screws.

The Ranger spacecraft is attached to the adapter at Station 232.50 at 6 points. Three of the attach fittings carry shear and compression loads only and provide for the installation of the spacecraft separation springs. The other three attach points secure the spacecraft to the adapter by means of pin-puller latch mechanisms. Pyrotechnic actuation of the pin-puller mechanisms allows the spacecraft to separate from the adapter at an approximate relative separation velocity of 19 inches per second. The maximum spacecraft tip-off rate is 2.49 degrees per second.

*Station numbers are referenced to Agena B.

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Table 11-1

SEPARATION SYSTEM PERFORMANCE

Parameter	Design Tolerance	$\dot{\theta}$ (Deg/Sec)	α (Deg)
Spring Rate	± .23 lbs/in.	0.29	0.096
Spring Stroke	± .01 in.	0.11	0.036
Spring Precompression	± .01 in.	0.11	0.034
Spring Shaft Alignment	± .25 deg	0.08	0.120
Spacecraft C.G. Offset	± .10 in.	0.15	0.076
Agena C.G. Offset	± .04 in.	0.04	0.058
Residual Rate	± .112 deg/sec	0.11	0.432
V-Band Impulse Effect	± .10 deg/sec	0.09	0.368

$\dot{\theta}$ = Resultant spacecraft rotational rate.

α = Angular error in the spacecraft velocity vector due to separation.

Relative velocity — spacecraft to Agena = 33.2 in/sec.

Absolute spacecraft velocity = 23.8 in/sec.

Table 11-2

AGENA AND SPACECRAFT INERTIAL CHARACTERISTICS

	Agena	Spacecraft
Weight	1450 lb	570 lb
Roll Moment of Inertia	136.6 slug-ft^2	60 slug-ft^2
Pitch Moment of Inertia	1951.0 slug-ft^2	75 slug-ft^2
Yaw Moment of Inertia	1926.8 slug-ft^2	75 slug-ft^2
X C.G.	346.28 LMSC Sta.	20 in. from sep. plane
Y C.G. (Off-Set)	0.38 in.	1.5 in.
Z C.G. (Off-Set)	-0.04 in.	1.5 in.

Separation Plane = LMSC Sta. 238.6

Although not considered as an intrinsic part of the spacecraft adapter, contamination isolation of the spacecraft from the Agena forward rack is provided by an air-tight diaphragm consisting of magnesium pressure panels installed on the Agena at Station 244.6. A recessed fiberglass cup installed in the center bay completes the isolation provision and allows the spacecraft high-gain antenna to protrude into the Agena forward rack.

The spacecraft/Agena electrical interface is provided by two pyrotechnically operated spin-off electrical connectors. Vertical fiberglass shields are installed inboard of the connectors to prevent the connectors from striking the spacecraft at separation.

Figure 11-3 shows the Ranger spacecraft adapter configuration and lists dimensional tolerances. Figure 10-7 provides supplemental information in regard to critical dimensions and tolerances of the combined spacecraft shroud and adapter systems.

11.4 NIMBUS SPACECRAFT ADAPTER

The Nimbus spacecraft adapter is a fabricated conical assembly with machined magnesium rings at both ends. The adapter assembly extends from Station 244.50 to Station 220.506* and provides the transition from the forward ring of the Agena to the spacecraft interface. The outside diameters of the upper ring and lower ring are 57.12 inches and 60.02 inches, respectively. The rings are connected by a riveted assembly of 0.032-inch thick magnesium skin with eleven internal aluminum alloy longitudinal stiffener members and two external intermediate ring frames. Three 7-inch diameter access doors and four channel mounting brackets installed inside the forward ring for the spacecraft ejection actuators complete the structural assembly.

A segmented airtight diaphragm is installed at Station 244.5 on the Agena forward equipment rack. Two pneumatic check valves, one venting forward and

*Referenced to Agena B.

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one venting aft, provide Agena/shroud cavity air pressure equalization during ascent, and diaphragm structural protection in the event of Agena forward rack overpressure.

The adapter is attached to the Agena forward rack at Station 244.50 by means of eight 1/2 inch diameter bolts. The Nimbus spacecraft is attached to the adapter at Station 220.506 by installation of a V-band clamp system joined by pyrotechnically actuated cutter assemblies spaced 180 degrees apart.

At spacecraft separation, the V-band is released by detonation of the cutter assemblies. Eight spring and cable combination assemblies are attached to the adjacent exterior ring frame to restrain the V-bands at separation. Lanyards are threaded through brackets mounted on the V-band ends to anchor the severed assembly to the spacecraft adapter. The spacecraft is ejected from the adapter by the spring assemblies at a relative velocity of 5 ft/sec with a maximum tip-off rate of 1.0 deg/sec.

The Nimbus Agena/spacecraft electrical interface consists of two connectors at Station 244.50. One is connected to the Agena umbilical (J 100) and goes through the adapter to the spacecraft to monitor spacecraft instrumentation prior to separation. The second connector goes to the spacecraft separation pyrotechnics and also provides verification that separation has occurred. Both connectors go to the shroud separation pyrotechnics.

Figure 11-4 illustrates the Nimbus spacecraft adapter configuration and lists dimensions and tolerances. Figure 10-8 provides supplemental information regarding critical dimensions and tolerances of the combined spacecraft, shroud, and adapter systems.

11.5 ECHO II (COMSAT A-12) SPACECRAFT ADAPTER

The ECHO spacecraft adapter is a riveted conical assembly that is attached to the Agena forward rack at Station 244.50* and provides the transition from the interface ring to the spacecraft canister mounting interface at Station

*Referenced to Agena B.

262.50. The inside diameter of the upper ring is approximately 41.8 inches, and the outside diameter of the base ring is approximately 60.0 inches. The adapter is comprised of four ejection spring support assemblies riveted to a machined ring at the base and coupled together at the top with machined ring segments that form an integral ring concentric with the base diameter. Additional interior structural support is provided by intermediate sheet metal formed shear webs, fittings, brackets, and clips. The exterior sheet metal skin is reinforced with longitudinal stiffening angles and doublers. Installation of four adjustable ejection springs in the support assemblies completes the adapter.

The spacecraft adapter is attached to the interface adapter ring with sixteen 1/4 in. diameter bolts and barrel nuts. The spacecraft canister is attached to the adapter by installation of a V-band system joined by four equally spaced explosive bolt retainer assemblies. At spacecraft separation the V-band is released by detonation of the pyrotechnically actuated release assemblies. Lanyards attached to the release assemblies confine the whipping action of the separated bands. Ejection springs separate the canister from the Agena at a relative velocity of approximately 5 ft/sec.

Although not part of the spacecraft adapter, a pressure bulkhead is installed on the forward equipment rack at Station 244.5. A cutout or hole in the adapter and a rubber boot-type enclosure provides for a TV-camera monitor installation aft of the adapter, yet seals off the spacecraft area from the Agena forward rack area. A TV checkout lamp opening is located just above the TV camera opening and is sealed with plexiglass. A "stovepipe" duct vents aft through a propellant tank fairing to permit exhausting of air during ascent.

The Echo II/Agena electrical interface is provided by two connectors mounted at Station 244.50. One connector provides for launch pad monitoring of spacecraft temperatures through the Agena umbilical. It also provides for two spacecraft canister temperature and one pressure measurement during flight plus event occurrence of spacecraft separation. The second connector serves the spacecraft canister separation pyrotechnics. Both connectors go to the shroud separation pyrotechnics.

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Figure 11-5 illustrates the adapter configuration and lists dimensions and tolerances. Figure 10-9 indicates the manner in which the spacecraft canister is mated to the adapter and lists critical dimensions and tolerances of the overall shroud and adapter system.

11.6 OGO SPACECRAFT ADAPTER

The OGO spacecraft adapter is an A-frame or truss-type structure that extends from Station 250.50 to Station 231 on the EGO Agena B vehicles, and from Station 245 to Station 227.13 on the POGO Agena D vehicles. A separate interface ring attaches to the Standard Agena forward rack and provides for the attachment of the adapter. The functional designs of the spacecraft adapters are the same for POGO and EGO programs.

The adapters are comprised of four machined A-frame fittings bolted to the interface adapter ring and provide the transition from the forward ring of the Agena to the interface of the spacecraft. Each A-frame provides support and is an attachment point for a corner of the spacecraft. The upper faces of the A-frames are machined surfaces that form a level interface plane for mating the 30x32 inch end of the rectangular spacecraft to the adapter.

The spacecraft is attached to the adapter at Station 231.00 on EGO and 227.13 on POGO by the installation of a V-band system joined by two double ended explosive bolts installed in pairs and spaced 180 degrees apart which release the V-band at spacecraft separation. The V-band system is comprised of V-blocks installed on the uppermost portion of each A-frame, tension band adjustment assemblies mounted on opposite sides of the spacecraft base structure, and interconnecting band assemblies to provide the necessary clamping action. At the time of spacecraft separation, compression springs located adjacent to the inside radius of each V-block eject the spacecraft from the adapter at a relative separation velocity of 5.0 ± 0.5 feet per second, with a tip-off rate of less than one degree per second.

To prevent contamination of the POGO spacecraft, a pressure diaphragm equipped with pressure relief valves for spacecraft/shroud cavity venting is installed on the Agena D forward equipment rack at Station 247.00. The pressure diaphragm installed on the EGO Agena B forward equipment rack at Station 250.50 is functionally the same, except that it is constructed of sealed segments.

The EGO spacecraft/Agena electrical interface is provided by four diaphragm connectors. One is devoted to interstage instrumentation, one to the payload separation pyrotechnics, one to the payload video coaxial cable, and one to the spacecraft preflight disconnect.

The preflight rotary disconnect provides a signal path through the Agena and its umbilical (J100) to the blockhouse, and is pyrotechnically disconnected at the spacecraft/Agena interface just prior to launch. The spacecraft video coaxial cable goes through the Agena umbilical (J100) to the spacecraft monitoring console, and the other connectors are used during flight. Figure 11-6 shows a typical POGO adapter configuration, and provides locating dimensions for the various components.

11.7 OAO SPACECRAFT ADAPTER

The OAO spacecraft adapter is a riveted conical assembly designed to adapt the smaller Agena diameter to the larger diameter spacecraft. The adapter assembly, together with the interface ring, is bolted to the Agena forward rack at Station 247.0 and extends to the spacecraft separation plane at Station 232.25. Eight triangular shaped trusses riveted to a machined ring provide primary structural support for the spacecraft attachment points and ejection spring housings. The outside diameter of the spacecraft adapter base is approximately 60 inches and the outside diameter of the adapter at the spacecraft separation plane is approximately 80 inches. Aluminum alloy skin 0.071 inch thick is riveted to the truss and ring assembly. The skin is reinforced internally with longitudinal channel and angles. Installation of four adjustable ejection springs in the support assemblies completes the primary adapter.

The octagonally shaped OAO spacecraft is attached to the top of the adapter by installation of a Marmon clamp system joined by explosive latch retainer assemblies. The clamp system is comprised of four separate clamps and fittings installed on each alternate truss and held in position by band and retainer assemblies mounted on the adjacent trusses. Pyrotechnic actuation of the retaining latch assemblies releases the band and clamp assemblies to allow spacecraft separation. Lanyards restrain the bands and clamps and prevent them from contacting the spacecraft. The ejection springs provide a relative separation velocity of 5 ft/sec with a maximum angular rate of separation of less than 1 degree/sec about any axis.

A pressure dome mounted inside the adapter and attached to the forward ring provides for compartment isolation for contamination control. Isolation of the spacecraft compartment from the rest of the vehicle is completed by a diaphragm installed between the nose-fairing and adapter. This diaphragm consists of a honeycomb ring fastened to the interstage with a separable seal to another honeycomb ring fastened to the nose-fairing halves. This flexible seal allows the movement necessary during stackup, fairing growth during ascent, and fairing jettison. The spacecraft, adapter, nose-fairing, and diaphragm are assembled as an encapsulated unit for transport to the launch site and assembly with the Agena. A quick-disconnect coupling and line through the adapter provides spacecraft nitrogen purging capability.

The Agena/spacecraft adapter electrical interface is provided by six connectors. One is attached to a harness that is connected to the Agena umbilical and provides a signal path through the adapter to the spacecraft to monitor prelaunch conditions. Three of the connectors are devoted to interstage and spacecraft instrumentation during ascent and prior to spacecraft separation. The two remaining connectors are devoted to interstage and spacecraft pyrotechnic functions. The functions include panel jettison, boom extension, and solar array panel extension, as well as spacecraft separation.

Figure 11-7 illustrates the OAO spacecraft configuration and provides dimensions and tolerances of components of the assembly. Supplemental information for the composite assembly of the shroud and adapter is provided in Fig. 10-12.

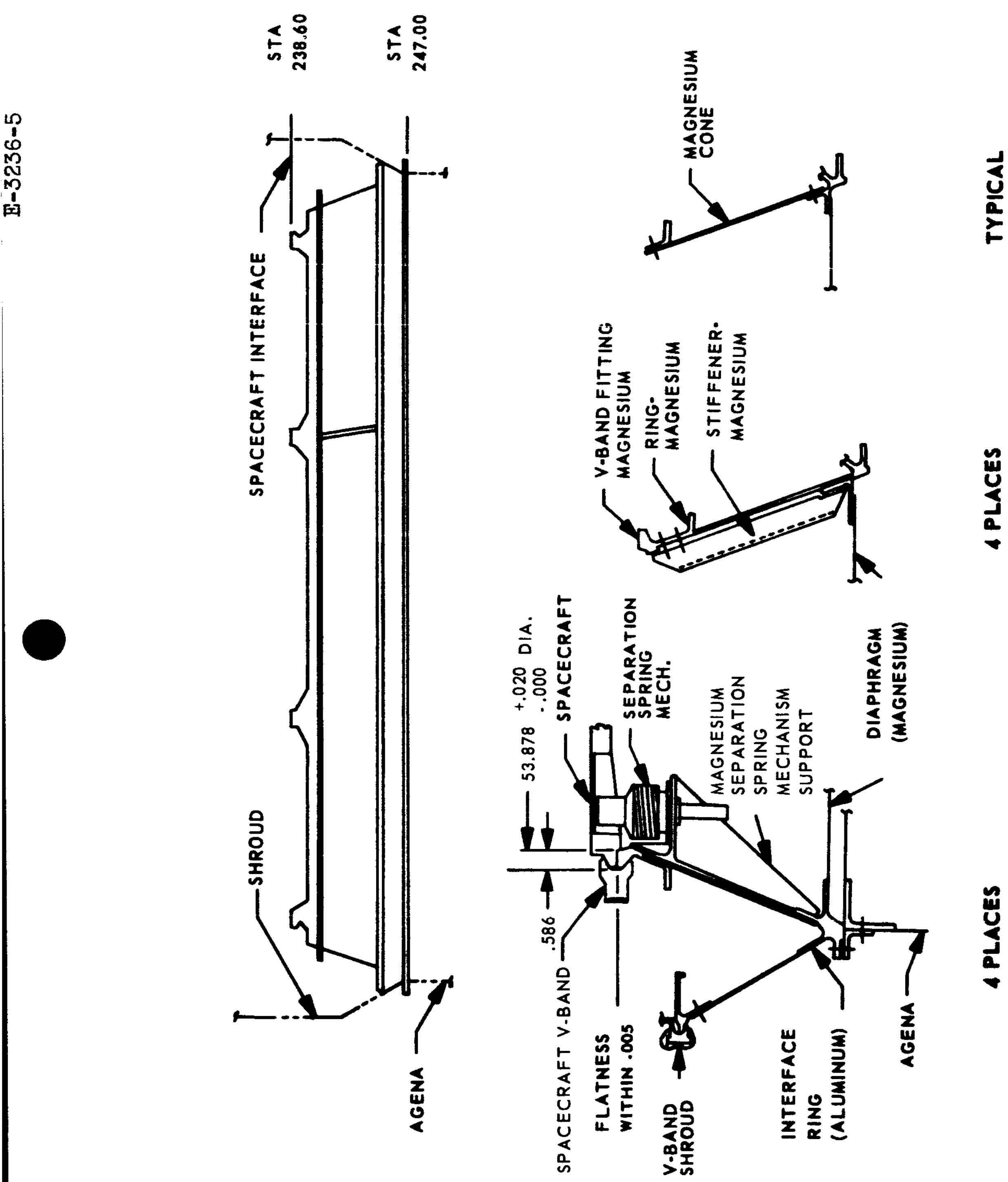

Figure 11-1 Mariner Mars Spacecraft Adapter

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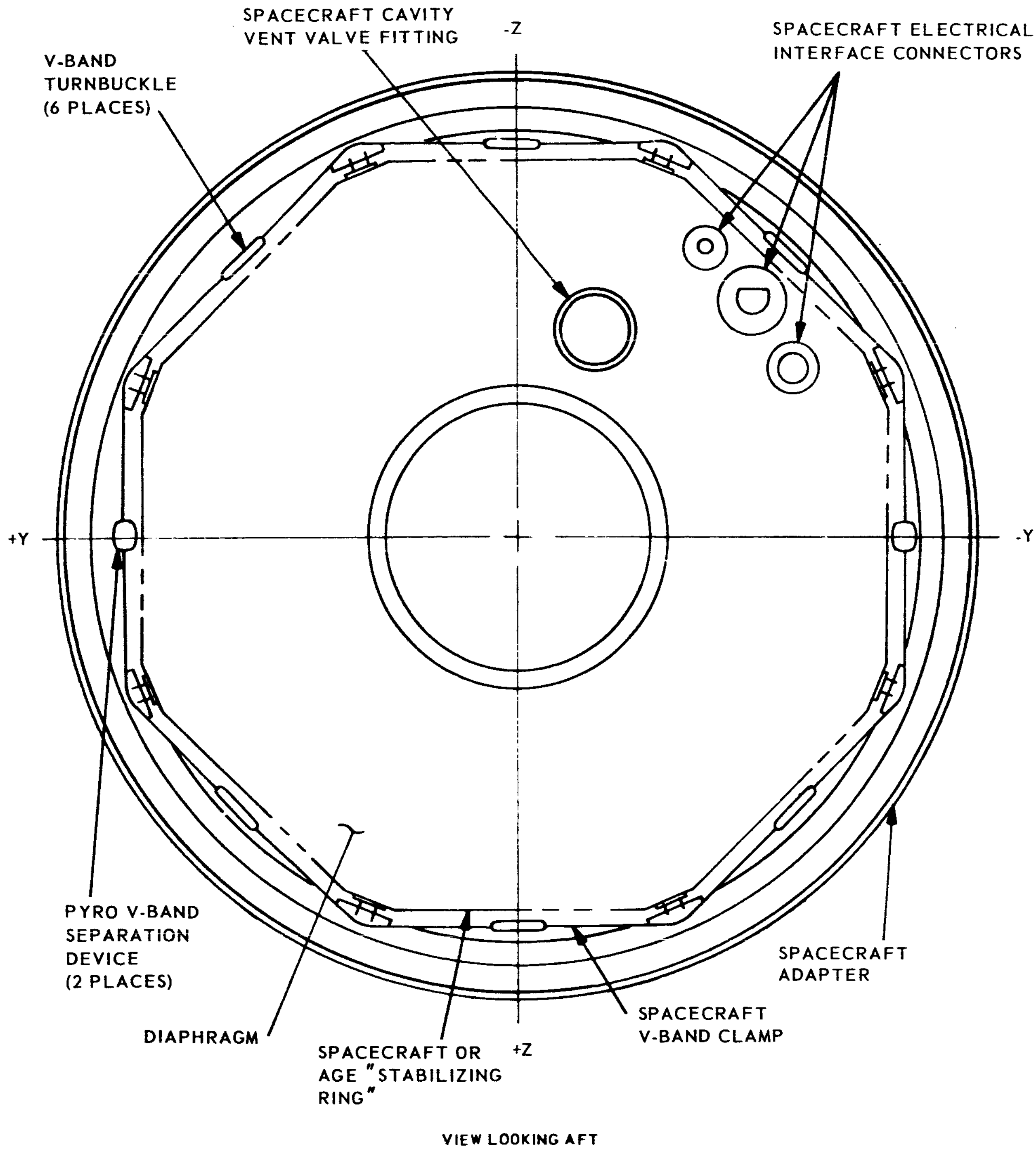
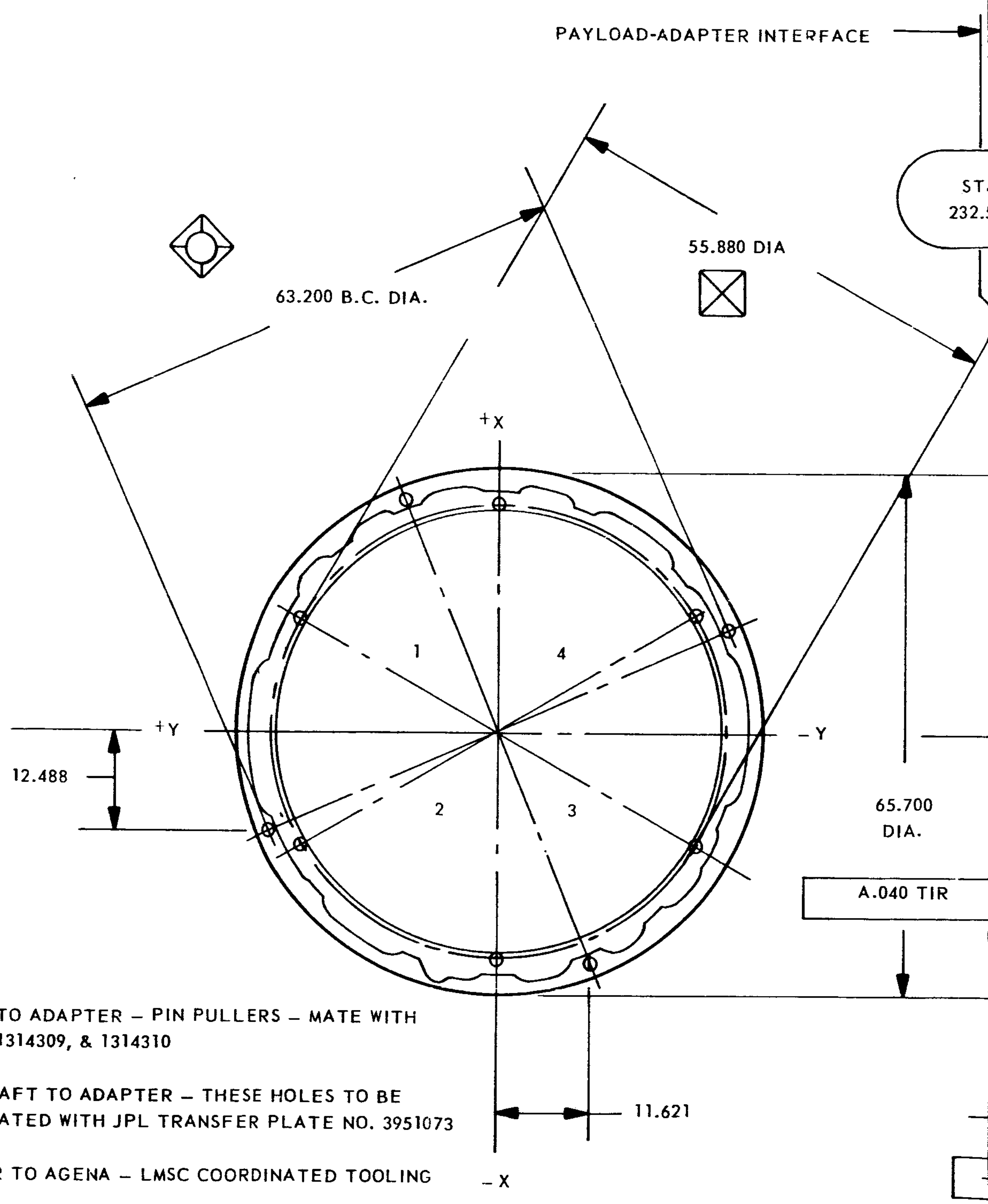


Fig. 11-2 Mariner Mars Spacecraft V-Band Clamp Assembly Installation



SHROUD TO ADAPTER – PIN PULLERS – MATE WITH 1314323, 1314309, & 1314310

SPACECRAFT TO ADAPTER – THESE HOLES TO BE COORDINATED WITH JPL TRANSFER PLATE NO. 3951073

ADAPTER TO AGENA – LMSC COORDINATED TOOLING

VIEW LOOKING AFT

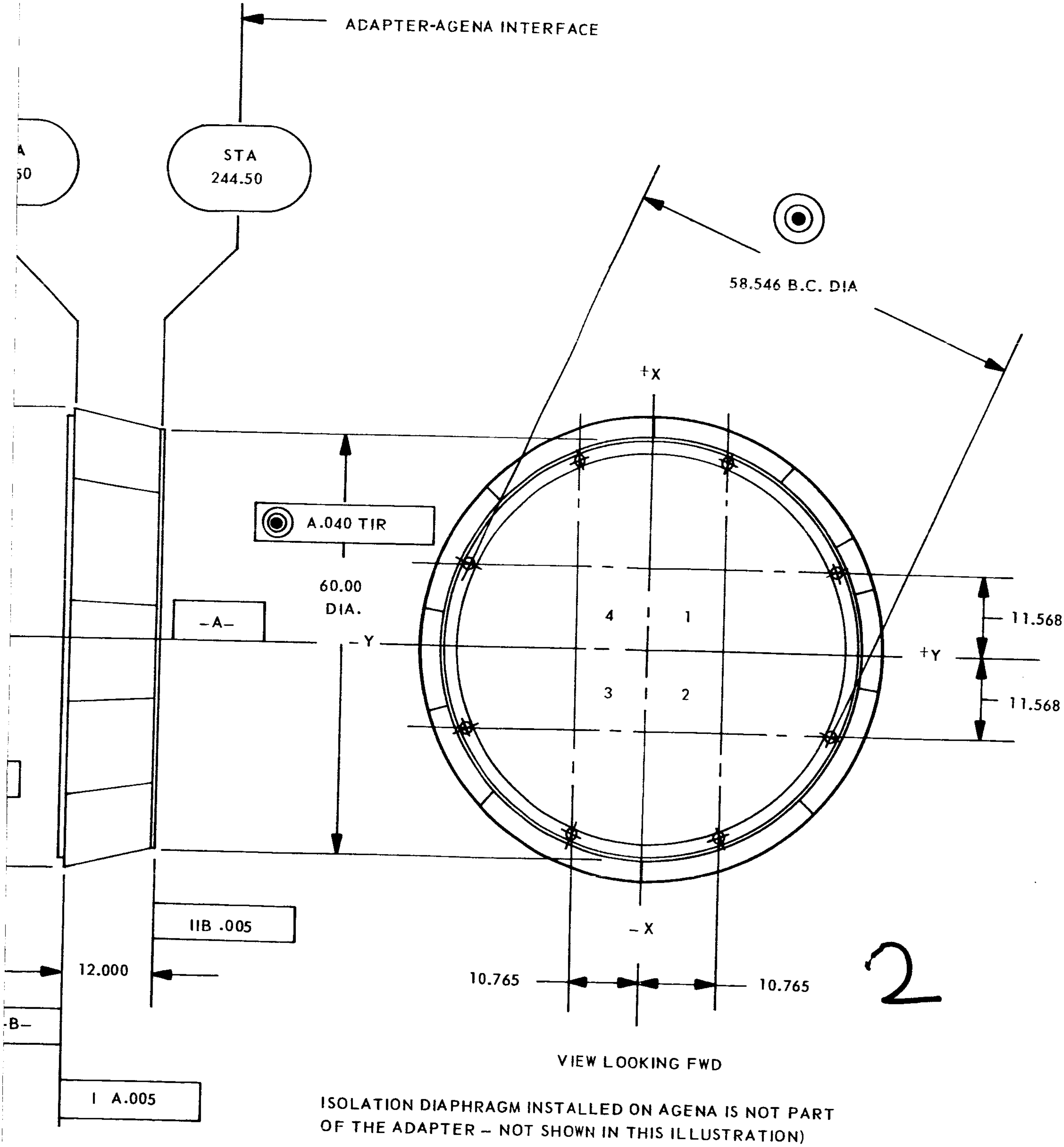
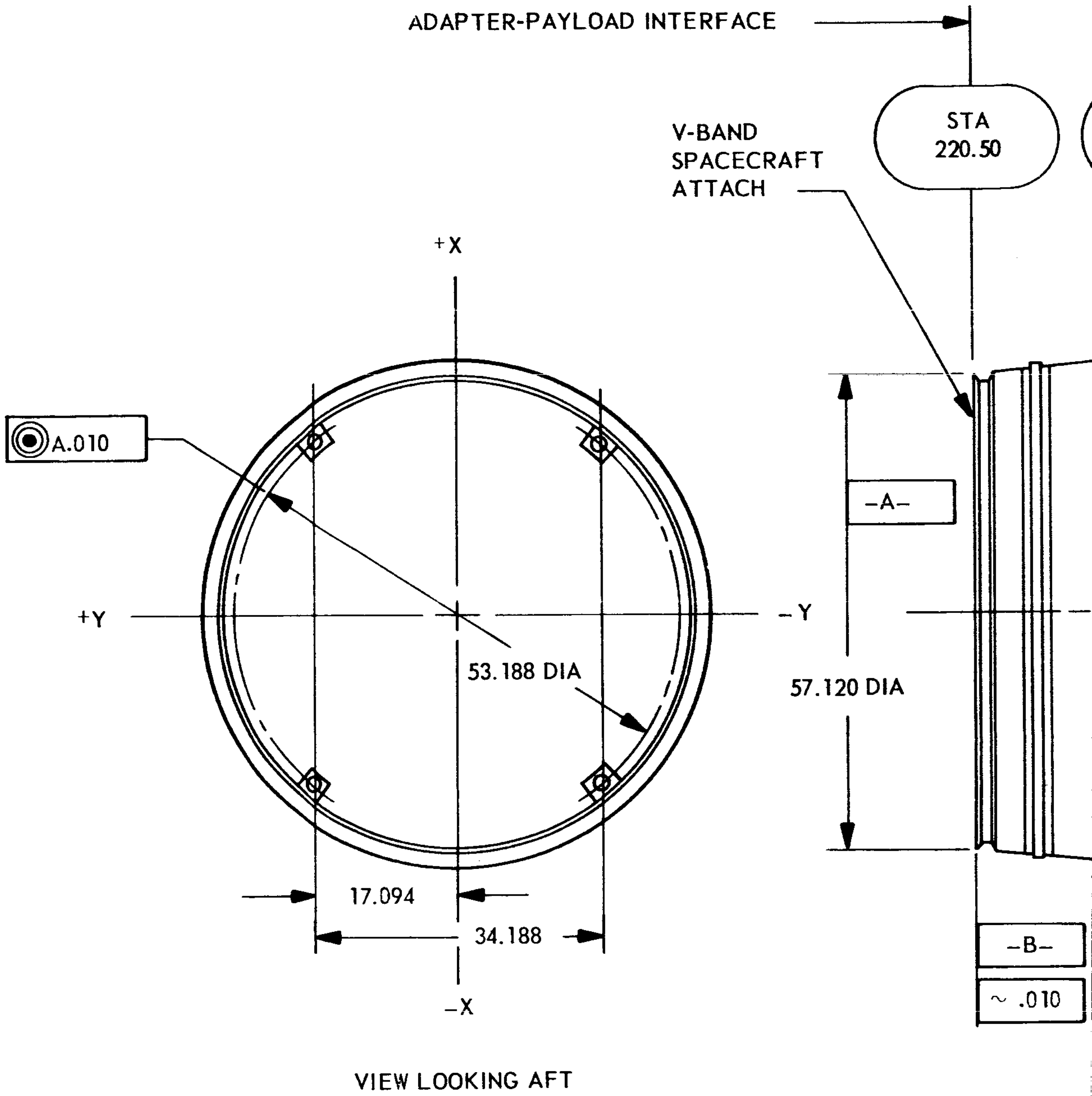


Fig. 11-3 Ranger Spacecraft Adapter



ADAPTER TO AGENA – LMSC COORDINATED TOOLING

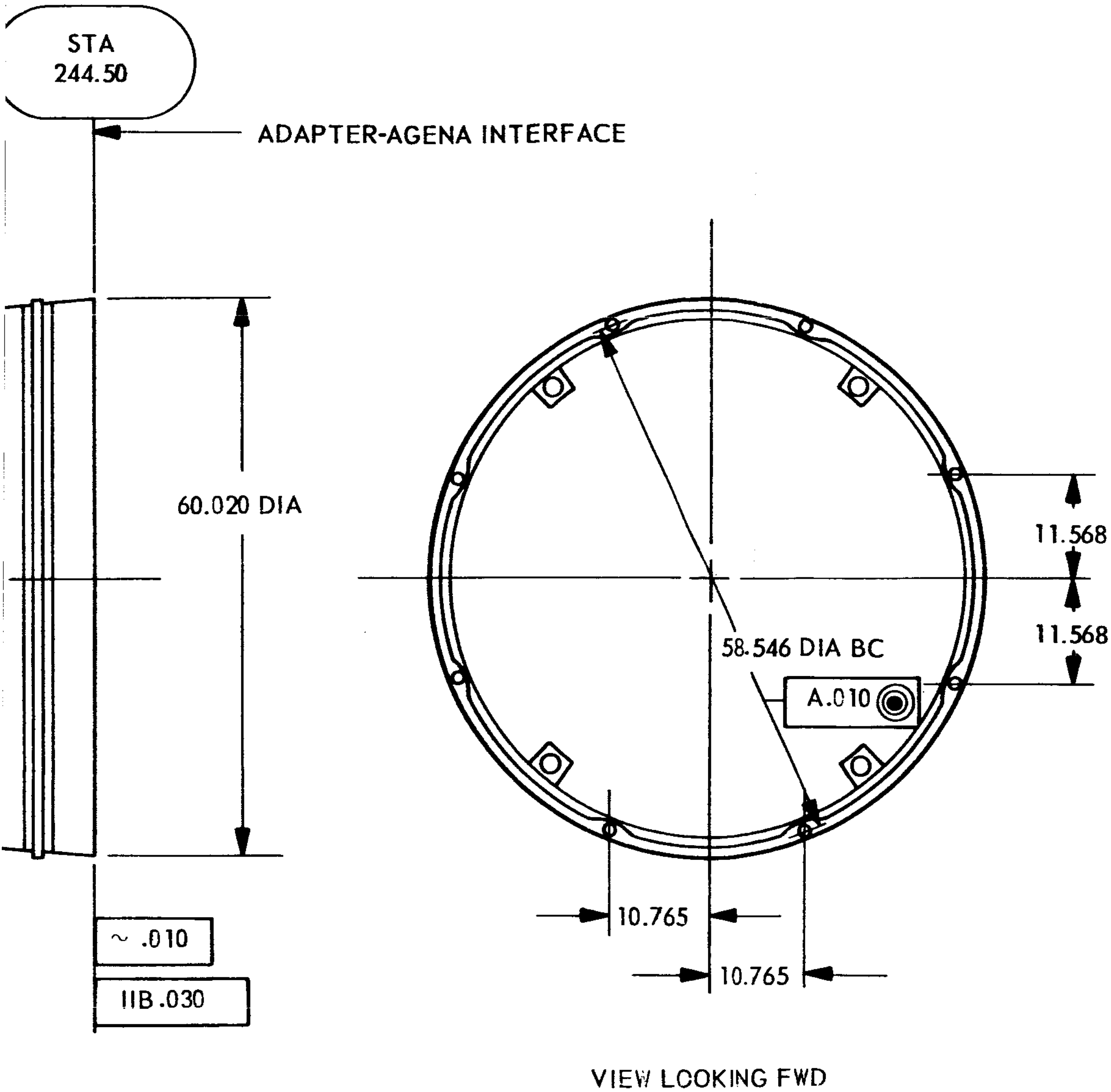


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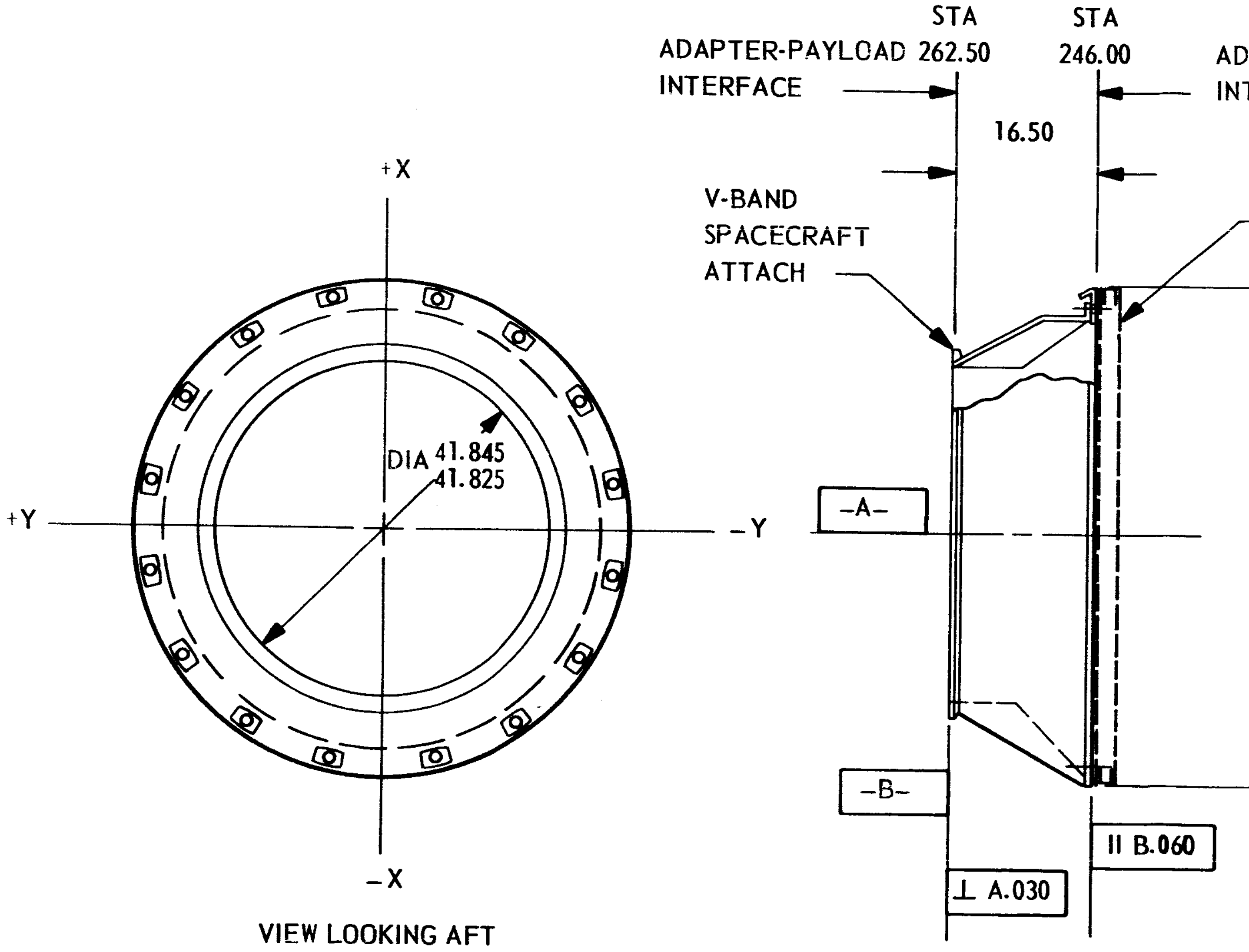
Fig. 11-4 Nimbus Spacecraft Adapter

E-3236-5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INTERFACE
STA
246.00
AD
INT
16.50
V-BAND
SPACECRAFT
ATTACH
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41.825
+Y
-Y
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II B.060
⊥ A.030
-X
VIEW LOOKING AFT

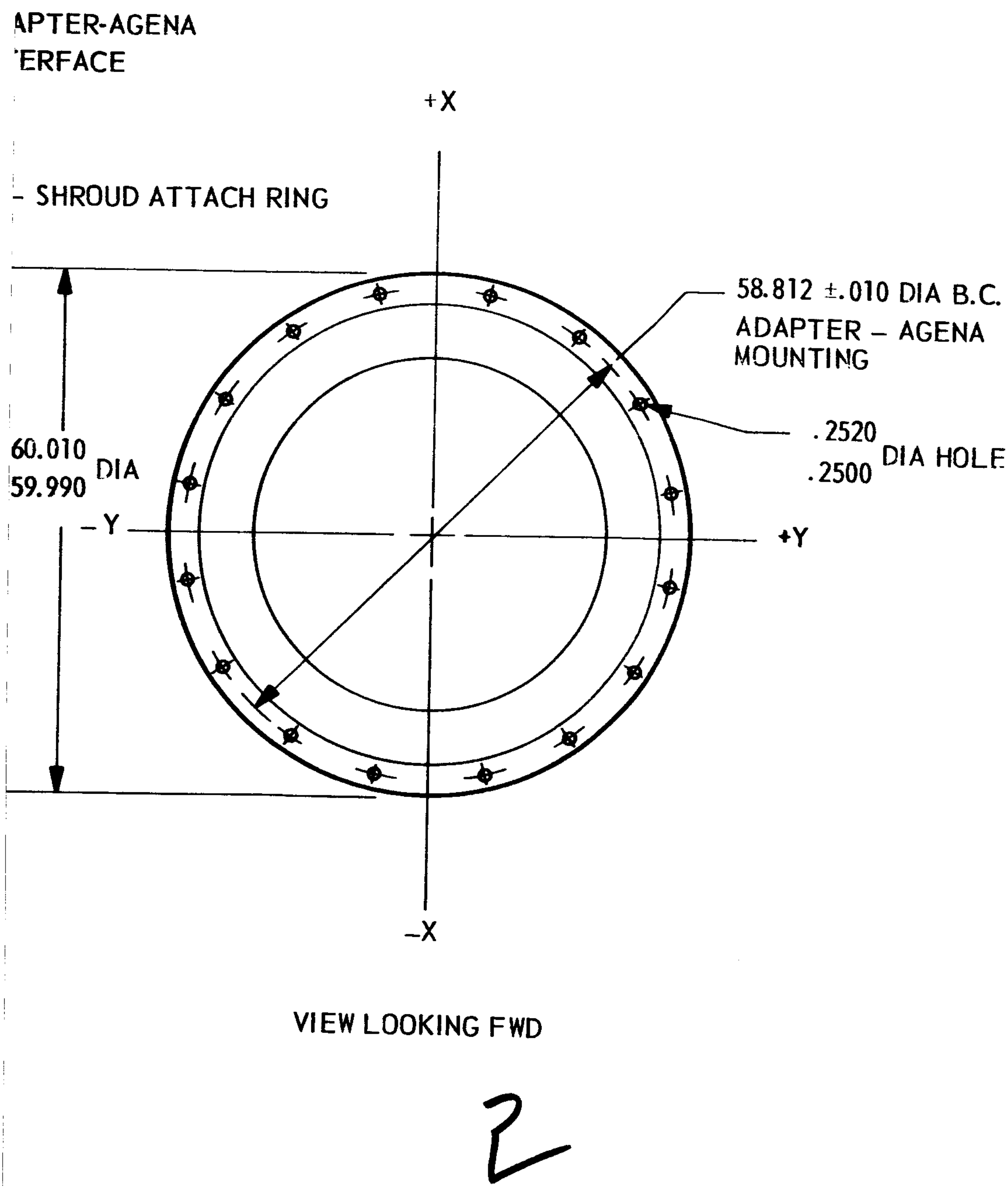
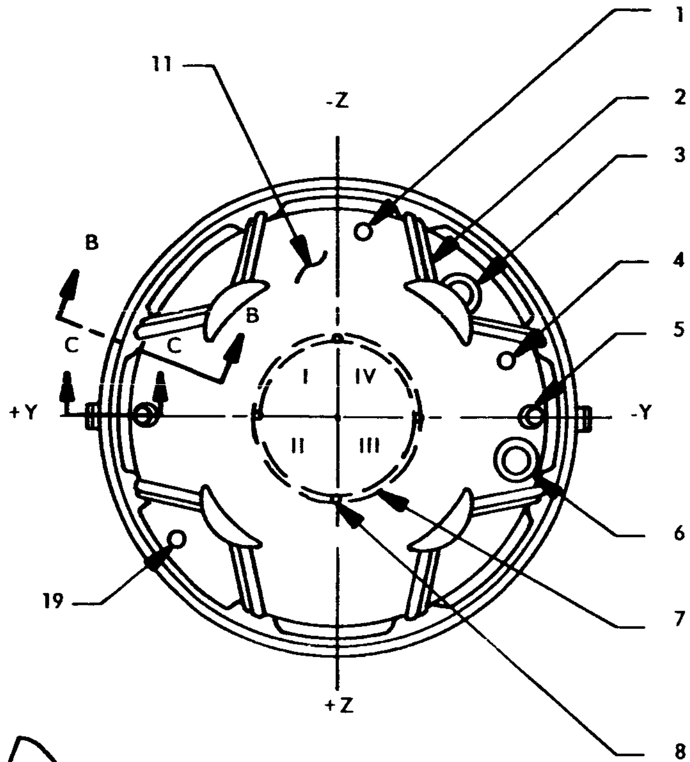
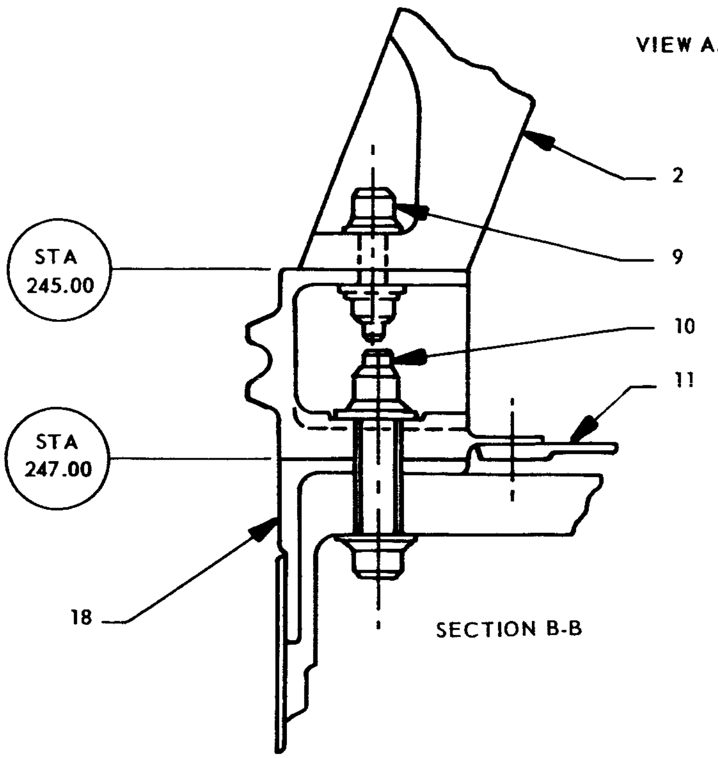


Fig. 11-5 Echo II Spacecraft Adapter

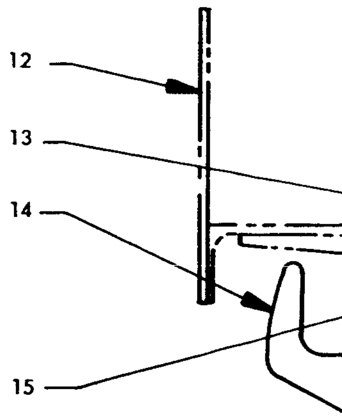
E-3236-5



VIEW AA



SECTION B-B



SECTION C-C

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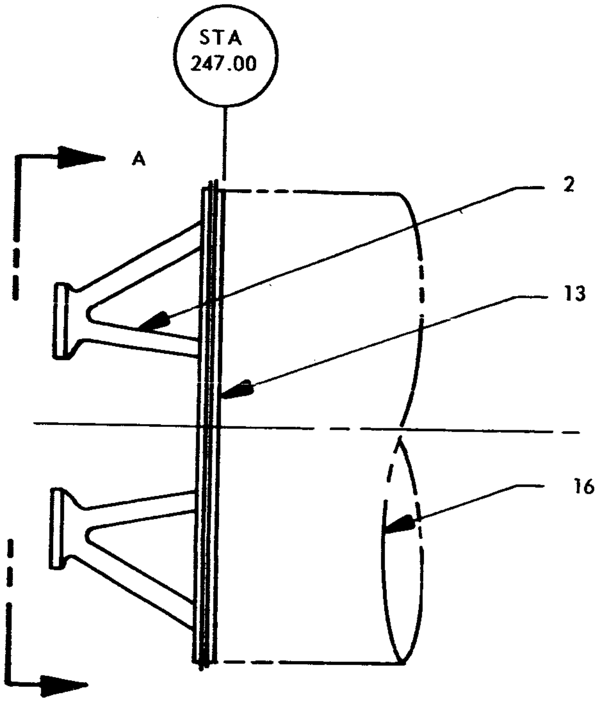

LEGEND

1. DIAPHRAGM CONNECTOR – TO TLM HARNESS
2. INTERSTAGE ASSEMBLY (STL)
3. PRESSURE RELIEF VALVE (FWD)
4. DIAPHRAGM CONNECTOR – TO SPACECRAFT SEPARATION PYROTECHNIC HARNESS
5. SHROUD LANYARD DISCONNECT PLUG (2)
6. PRESSURE RELIEF VALVE (AFT)
7. "Z" RING
8. COLUMN SUPPORTS – DIAPHRAGM (4)
9. BOLT – INTERSTAGE (8)
10. BOLT – ADAPTER RING (8)
11. DIAPHRAGM
12. SHROUD (DAC)
13. ADAPTER RING
14. PIVOT BRACKET (AFT) (2)
15. PIVOT BRACKET (FWD) (2)
16. AGENA – FWD MIDBODY
17. TORQUE – BOX – PIVOT BRACKET (2)
18. MIDBODY FLANGE (FWD)
19. DIAPHRAGM CONNECTOR – TO SPACECRAFT PREFLIGHT PYROTECHNIC DISCONNECT HARNESS

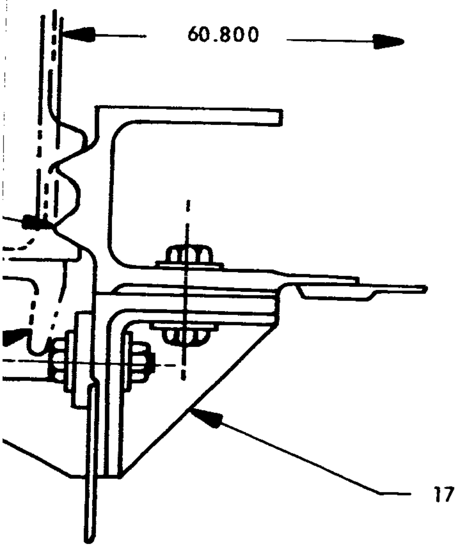

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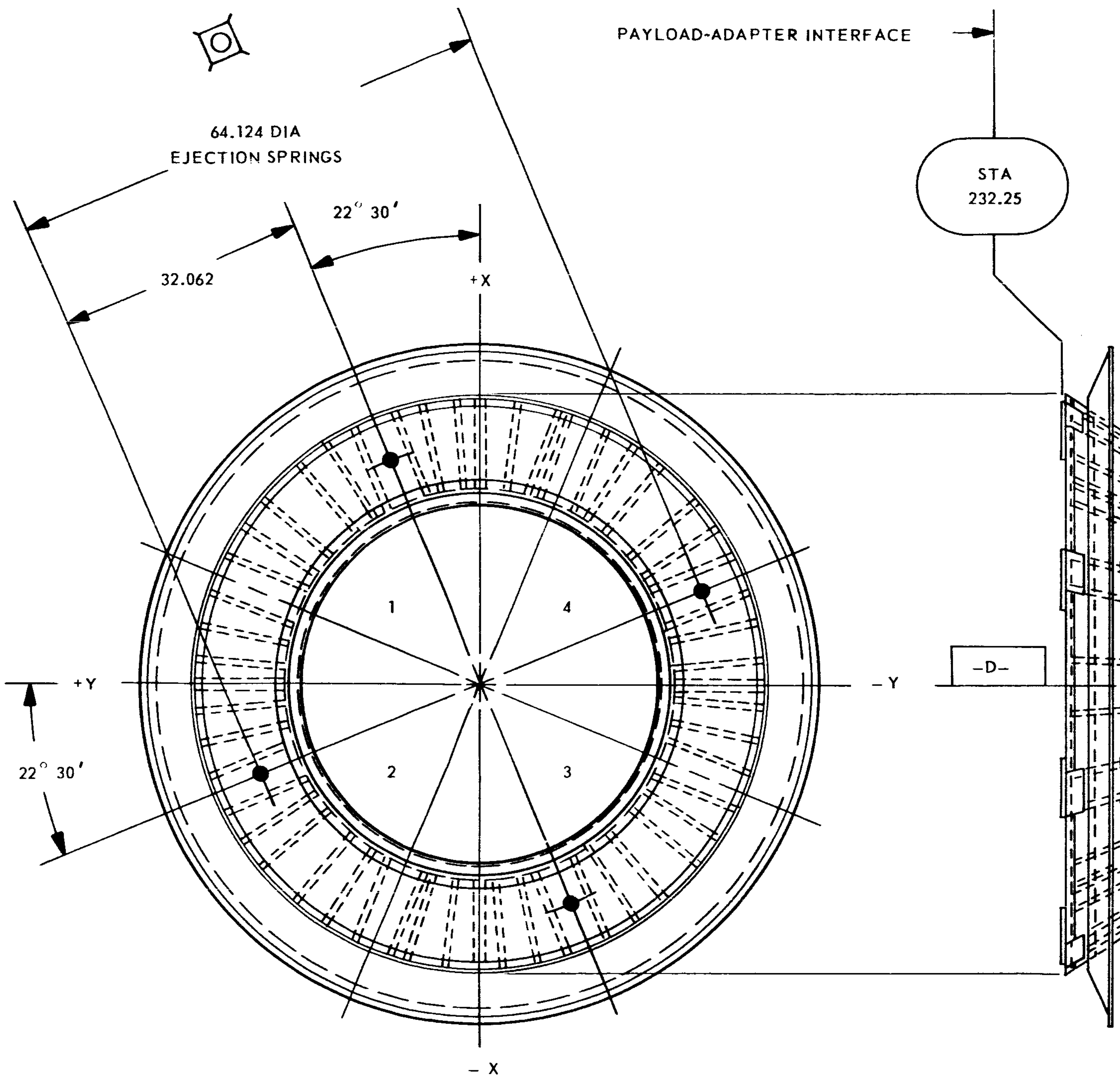
Fig. 11-6 POGO Adapter Configuration



VIEW LOOKING AFT

SPACECRAFT TO ADAPTER – GAEC COORDINATED TOOLING

ADAPTER TO AGENA – LMSC COORDINATED TOOLING

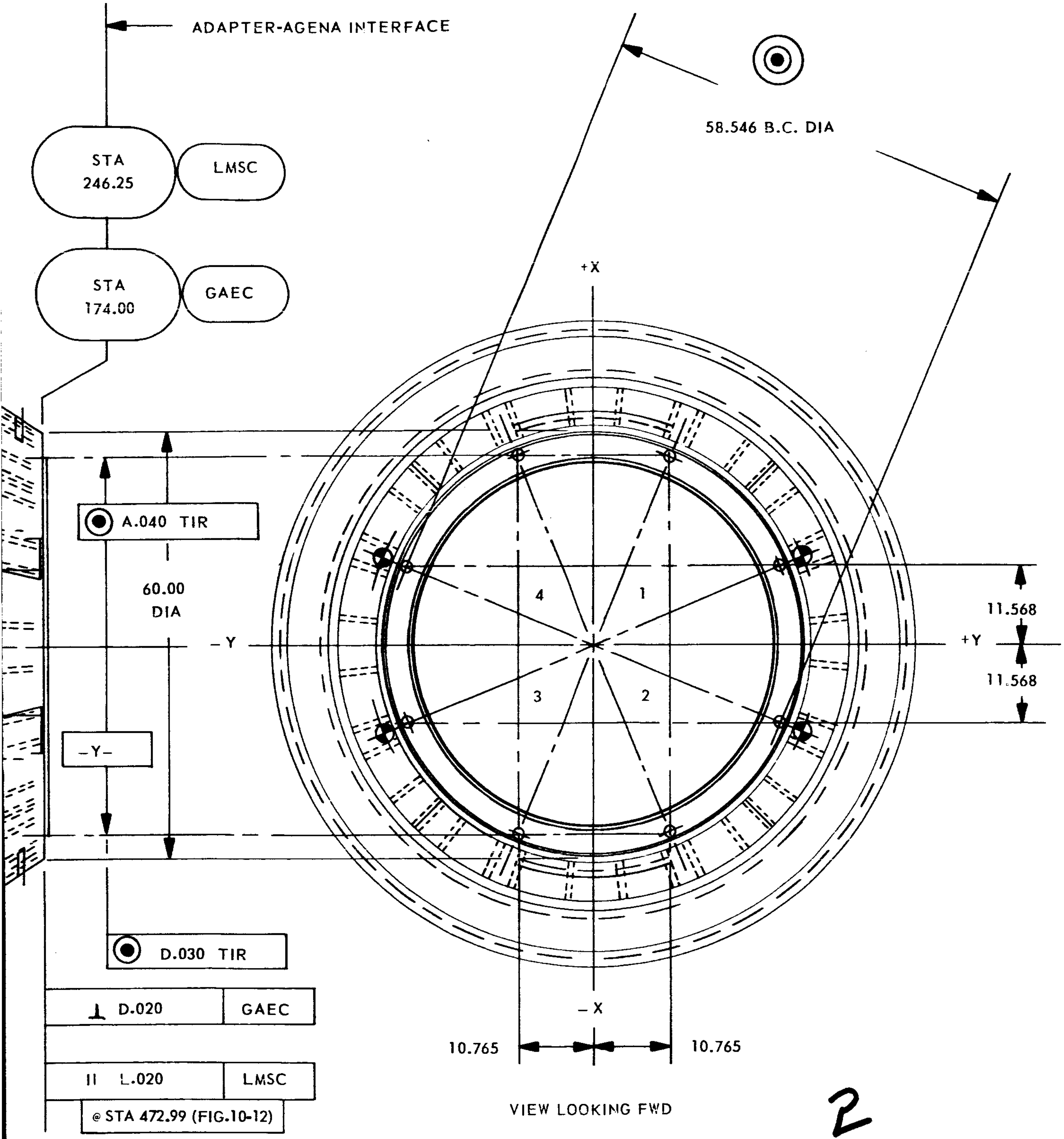


Fig. 11-7 OAO Spacecraft Adapter

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SECTION 12
OTHER SPACECRAFT OR MISSION SUPPORT SYSTEMS

12.1 GENERAL

In addition to the spacecraft support systems that have been discussed in Sections 10 and 11, program-peculiar systems may be provided that can be of potential value to the spacecraft designer. For example, the TV camera system flown on past missions is used to monitor spacecraft separation and post-separation functions. This system is discussed in the following paragraphs.

12.2 TV CAMERA SPACECRAFT SEPARATION MONITORING SYSTEM PROVISIONS

The Agena D vehicle can accommodate a television camera system to monitor spacecraft separation and post-separation functions. A GFE Lear-Siegler TV system was successfully employed on the Echo II mission.

Installation of the camera in the Echo II mission was in the forward equipment rack (except for the antenna); however, location forward in the adapter section may be feasible for new missions. The unit used for Echo II is independent of vehicle functions and has a self-contained power supply. Figure 12-1 illustrates the major components and their locations.

The TV system block diagram is shown in Fig. 12-2. Functional characteristics and requirements are summarized as follows:

Manufacturer (video equipment)	Lear Siegler, Inc.
Camera dimensions	12 x 2-1/2 x 2-1/2 in.
Lens	50 mm f/1.5 Wollensak
Lens filter	Wratten 47B (heavy blue filter)

Lens field of view	$9 \times 16^{\circ}$
Camera control dimensions	8 x 4 x 5-1/2 in.
Frame rate	30 frames/sec
Field rate	60 fields/sec (interlaced 2 to 1)
Horizontal resolution (real time)	525 TV lines
Video bandwidth required	4 Mc minimum
RF frequency	255 Mc
RF bandwidth	8 Mc min, 10 Mc max
Modulation	FM
RF power output	50 w min to 50-ohm load
Camera and camera control power supply	18, Yarnery Hr-18 Silvercells to deliver 400 amp
Antenna	2 bladed antenna array (each blade is bent quarter wave antenna)
Total airborne system weight	Approximately 67 lb

The length of time that the spacecraft can be viewed is dependent upon the lens field of view and relative motion between the spacecraft and Agena following separation. Normal Agena deadband control limits permitted observation of the Echo balloon through a 9 x 16 degree field of view lens for a period of approximately 10 minutes.

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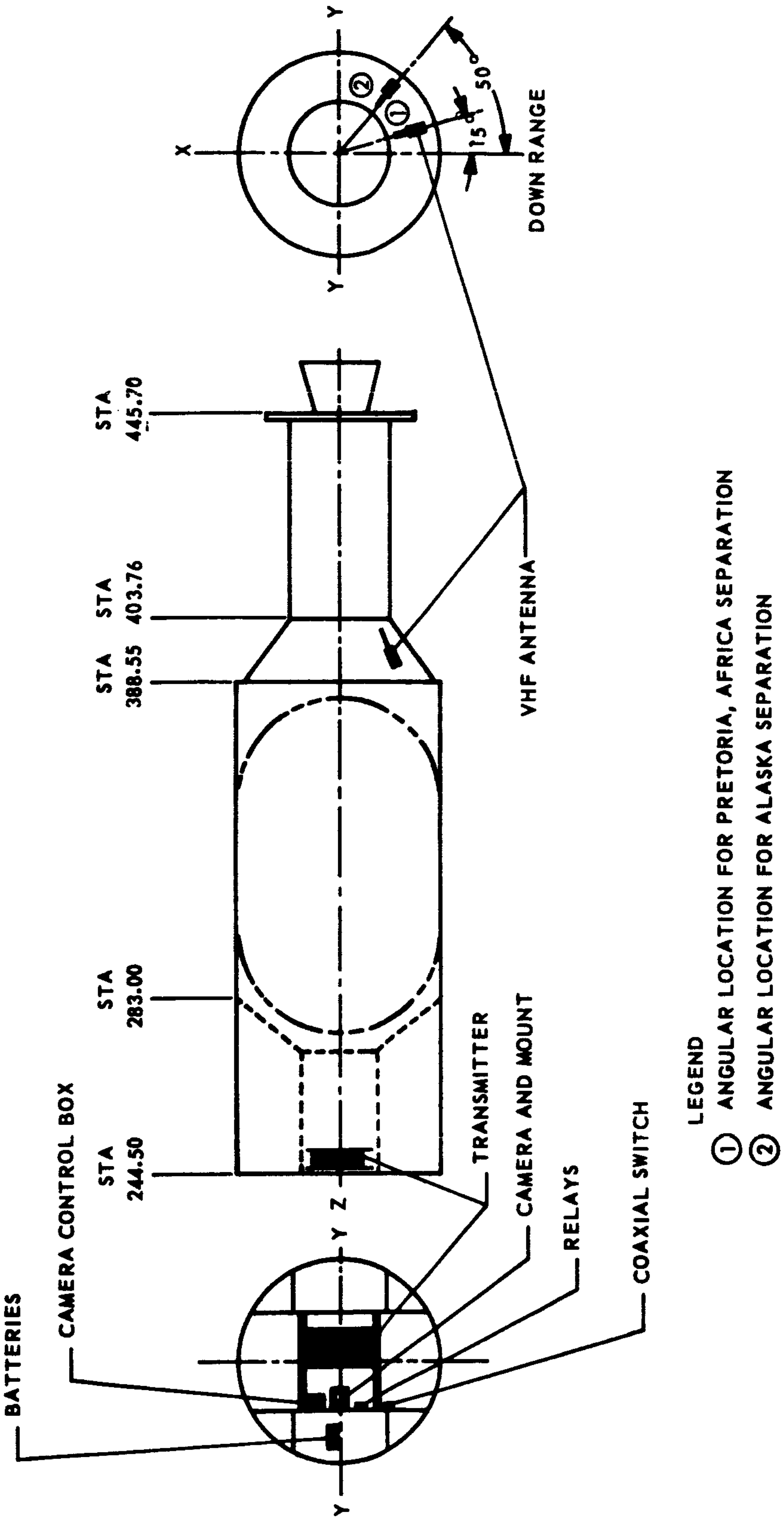

Fig. 12-1 TV Camera System Installation

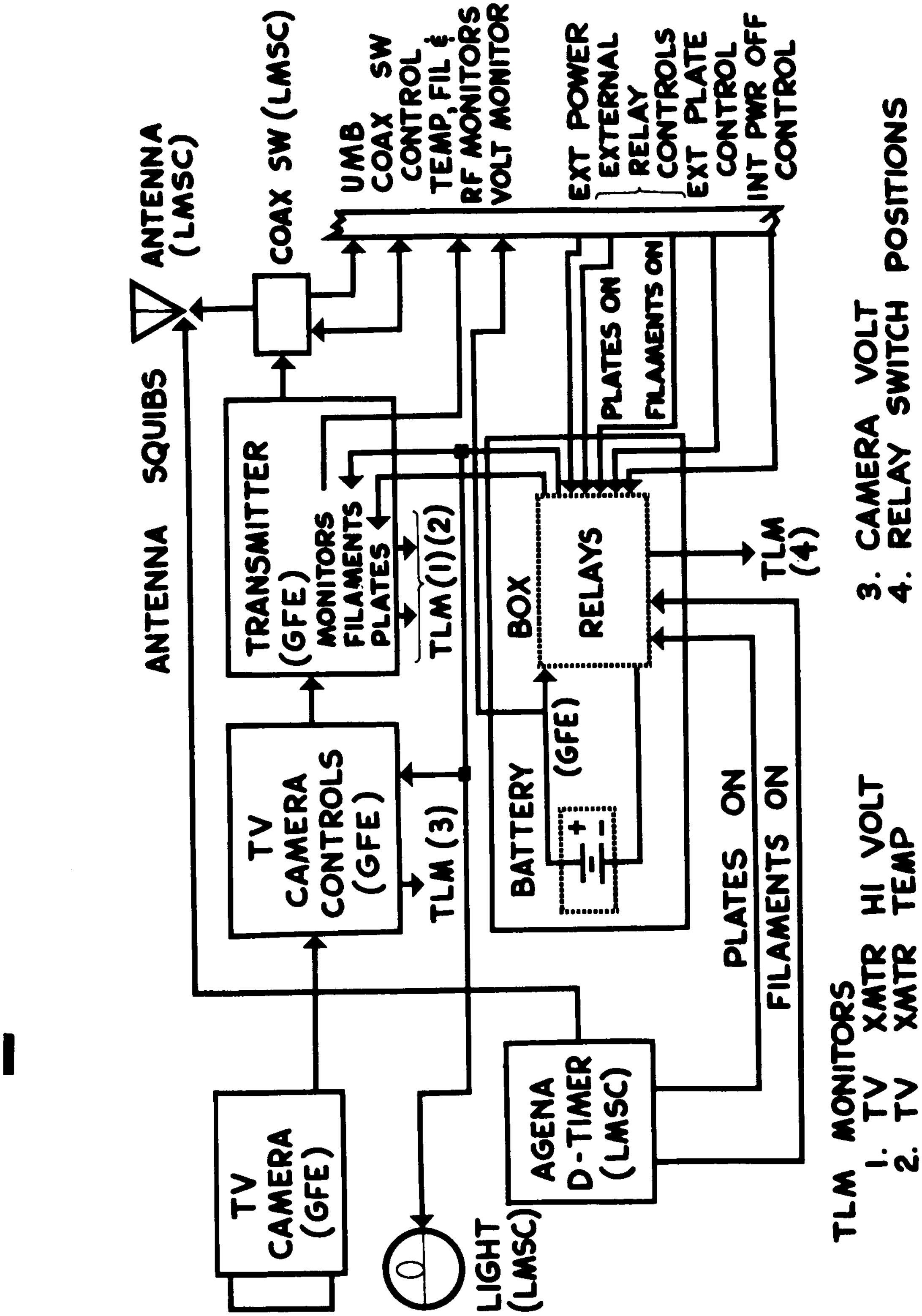

Fig. 12-2 TV Camera System Block Diagram (Echo II 6301 Application)

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SECTION 13
AGENA SPACEFRAME SUBSYSTEM

13.1 GENERAL

The Agena D spaceframe subsystem consists of three major vehicle structural sections: the forward rack, the tank section, and the aft rack. These sections provide for the mounting and installation of the basic, optional, and peculiar equipment that makes up the propulsion, electrical, guidance, and communications subsystems. A fourth major spaceframe section is the booster adapter which provides for mating of the Agena to the particular booster being employed for that mission.

The entire spaceframe, excluding the booster adapter, extends from LMSC Station 247 to 462.5. Station 247 is at the plane between the spacecraft support hardware (mission peculiar shroud, adapter, etc., as discussed in Sections 11 and 12) and the basic vehicle forward section. The Agena/booster mechanical interface is at LMSC Station 492.21 (TAT) or 526 (Atlas). The overall length of the spaceframe aft of Station 247 is approximately 20 feet. Figure 13-1 shows the Agena D with the booster adapter partially removed and provides an indication of how the spaceframe serves the remaining subsystems in providing mounting and installation facilities for the necessary equipment. An inboard profile of a typical application (POGO) is presented in Fig. 13-2.

The spaceframe subsystem is made up primarily of basic hardware with a limited number of optional and peculiar items of equipment available for application on a prospective spacecraft mission. Bracketry and installation provisions for basic and optional equipment also form a part of the basic spaceframe hardware.

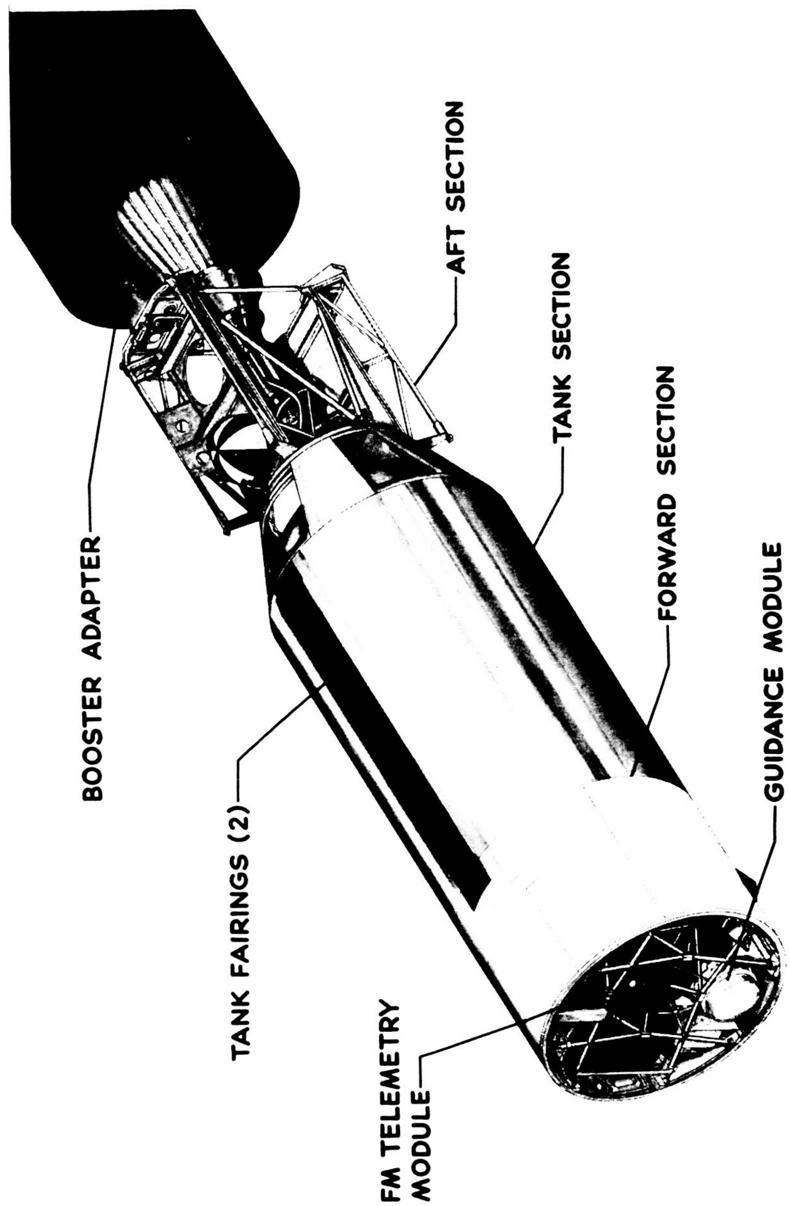
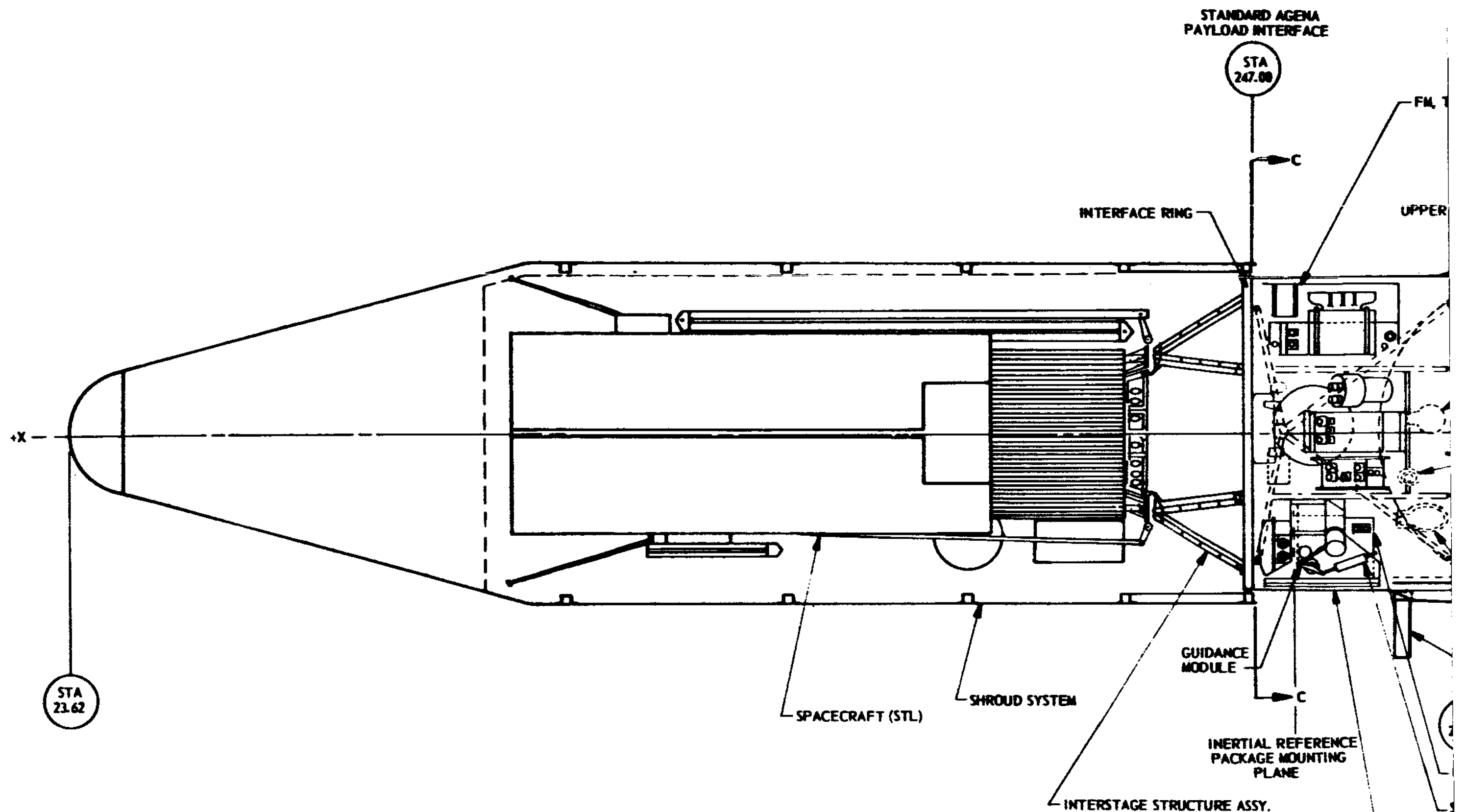
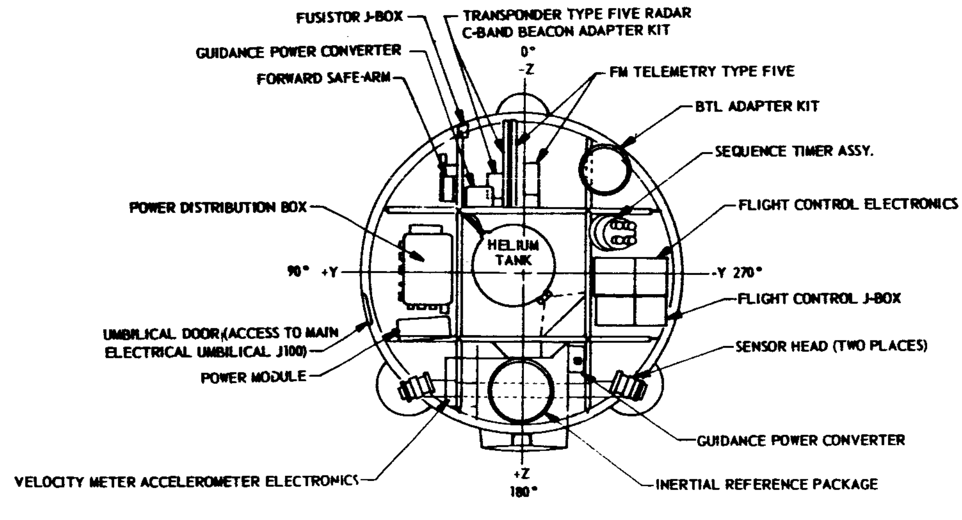


Figure 13-1 Agena D Basic Vehicle Spaceframe Sections

E-3236-5





VIEW C-C

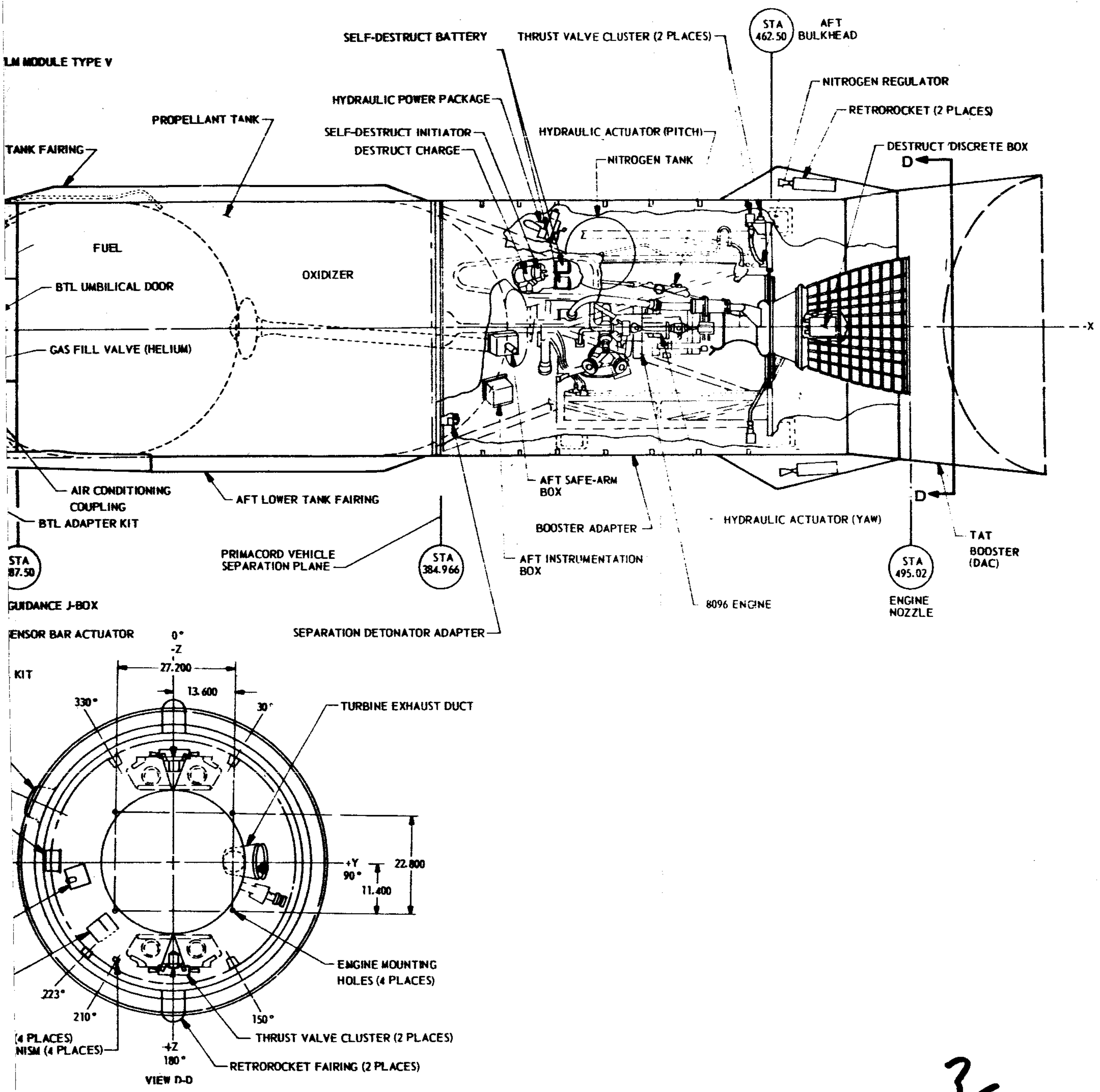


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Fig. 13-2 Typical Agena D Vehicle Application (POGO)

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13.2 BASIC HARDWARE

13.2.1 Forward Section

The forward section houses basic Agena system components (Fig. 13-3). Space and mounting is also provided for certain mission optional components and kits. The section is completely enclosed and is 40.5 inches long, extending from Station 247 to Station 287.5.

The forward section structure is a welded truss-type tubular aluminum frame, to which are bolted external rings, longerons, skin assemblies and access doors. Mounting brackets and shelves are provided within the interior on the tubular frame for the installation of components. Components may also be attached to the tubular frame directly by means of rubber-covered clamps.

Removable beryllium panels provide the principal access to components in the forward section. Additional access, when necessary, can be obtained by removal of beryllium fixed skins, which are bolted to rings and longitudinal members. With equipment installed, the vehicle cannot be erected or transported without flight or dummy panels in place.

The basic Agena committed strength at Station 247 is shown in Fig. 13-4. Additional information is available in detail specification LMSC-1414870. The locations of eight 0.5-inch diameter holes to be used for attaching program peculiar structures (e.g., spacecraft adapter, auxiliary rack, etc.) are shown in Fig. 13-5. The radial dimensions given are computed from measurements taken from LMSC Master Tool 1339773-501-14E-1. (Mating structures should be drilled from tools patterned from the master.) The four hardpoints for mounting a program-designed pressure diaphragm are also shown on Fig. 13.5.

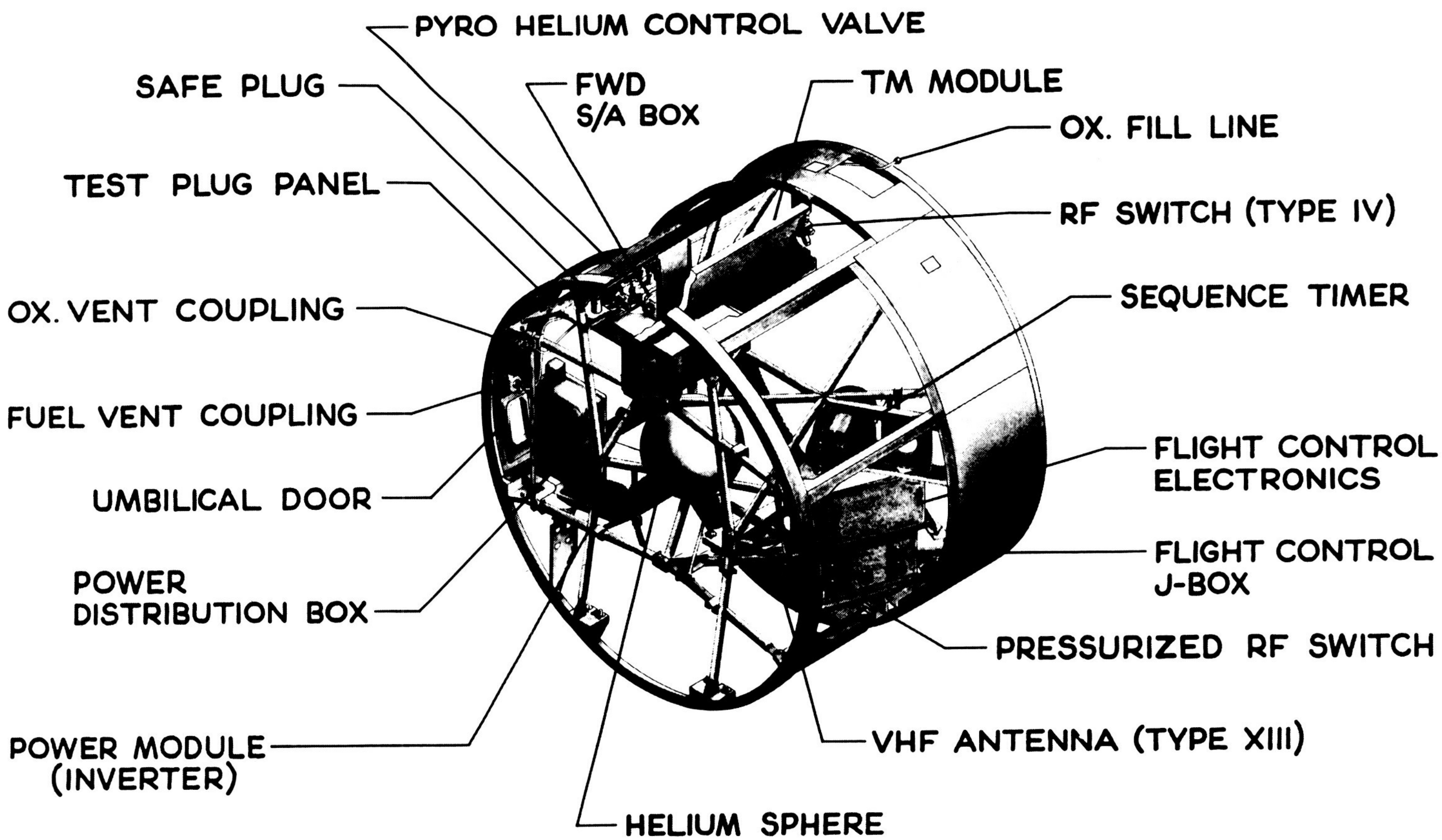


Figure 13-3 Forward Section, Basic Vehicle

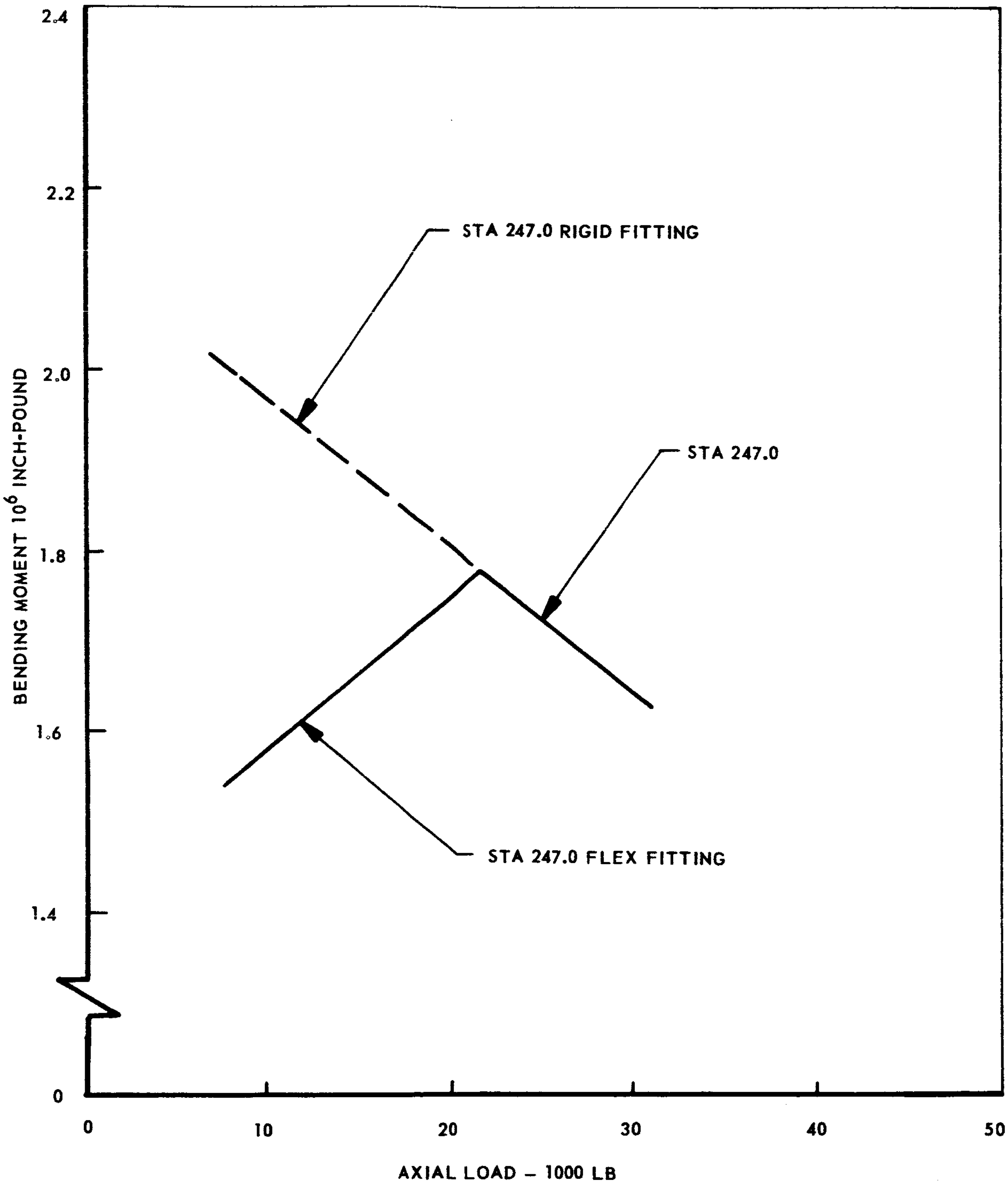
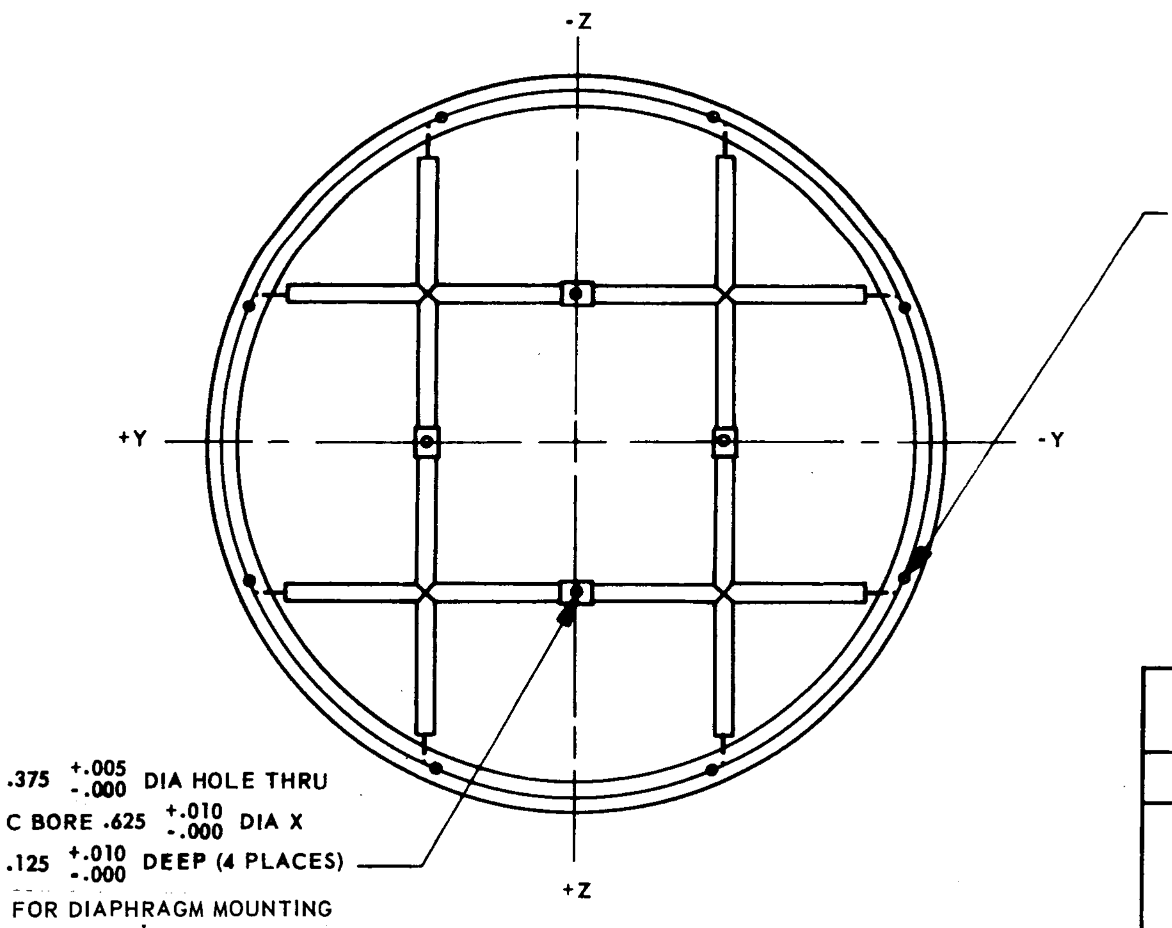


Figure 13-4 Standard Agena Committed Strength for Station 247



.5080 $^{+.0050}_{-.0000}$ DIA HOLE THRU
FOR PAYLOAD MOUNTING BOLTS
(8 PLACES)

RECOMMENDED BOLT FOR FULL STRENGTH:
DIAMETER .4995 $^{+.0000}_{-.0010}$ AFTER PLATING

HEAT TREAT 180–200,000 PSI UTS (MIN.)

STANDARD TORQUE 480–690 INCH-POUNDS

MATING BOLT HOLE LOCATIONS AT STA 247.0	
DEGREES (REF)	RADIUS (REF)
22°29'59"	29.006
67°30' 9"	29.009
112°29'55"	29.010
157°29'49"	29.010
202°29'33"	29.010
247°29'24"	29.009
292°29'33"	29.006
337°30' 3"	29.007

Figure 13-5 Payload and Diaphragm Mounting Hole Location

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13.2.2 Tank Section

The propellant tank section consists of a two-chamber assembly, each chamber having an attached passive fuel and oxidizer containment system or sump. The forward chamber is for fuel and the aft chamber for oxidizer; both chambers are separated by a common bulkhead. The sump outlet for the forward chamber is located within the aft chamber. The fuel supply line to the engine pump inlet passes through the oxidizer chamber as shown in Fig. 13-6.

The tank assembly (Fig. 13-6) is an integral part of the vehicle spaceframe and provides the supporting structure and exterior surface for the center portion of the vehicle. The assembly is constructed of aluminum sheet formed and welded together to make up the dual tank structure. Overall tank length, including the fore and aft hemispherical tank ends, is 129 inches. The minimum net volume of the fuel tank with baffles is 75.9 cubic feet, or a total of approximately 568 gallons. The minimum net volume of the oxidizer tank with baffles installed is 98.4 cubic feet, or approximately 736 gallons. Two external fairings mounted longitudinally along the tank section provide a passageway for the interconnecting plumbing and wiring between the forward and aft sections. In addition, the fairings serve as tunnels for transmitting air from the forward section to the aft section during on-pad air conditioning and ascent venting.

The propellant containment system (Fig. 13-6) provides propellant scavenging and passive propellant orientation in a sump located at the outlet of each tank. This arrangement ensures containment of sufficient propellants under zero gravity conditions to start and operate the main engine until the bulk of the propellants in the tanks become oriented over the pump inlets. The fuel sump has a nominal capacity of 0.25 cubic feet while the oxidizer sump has a capacity of 0.56 cubic feet.

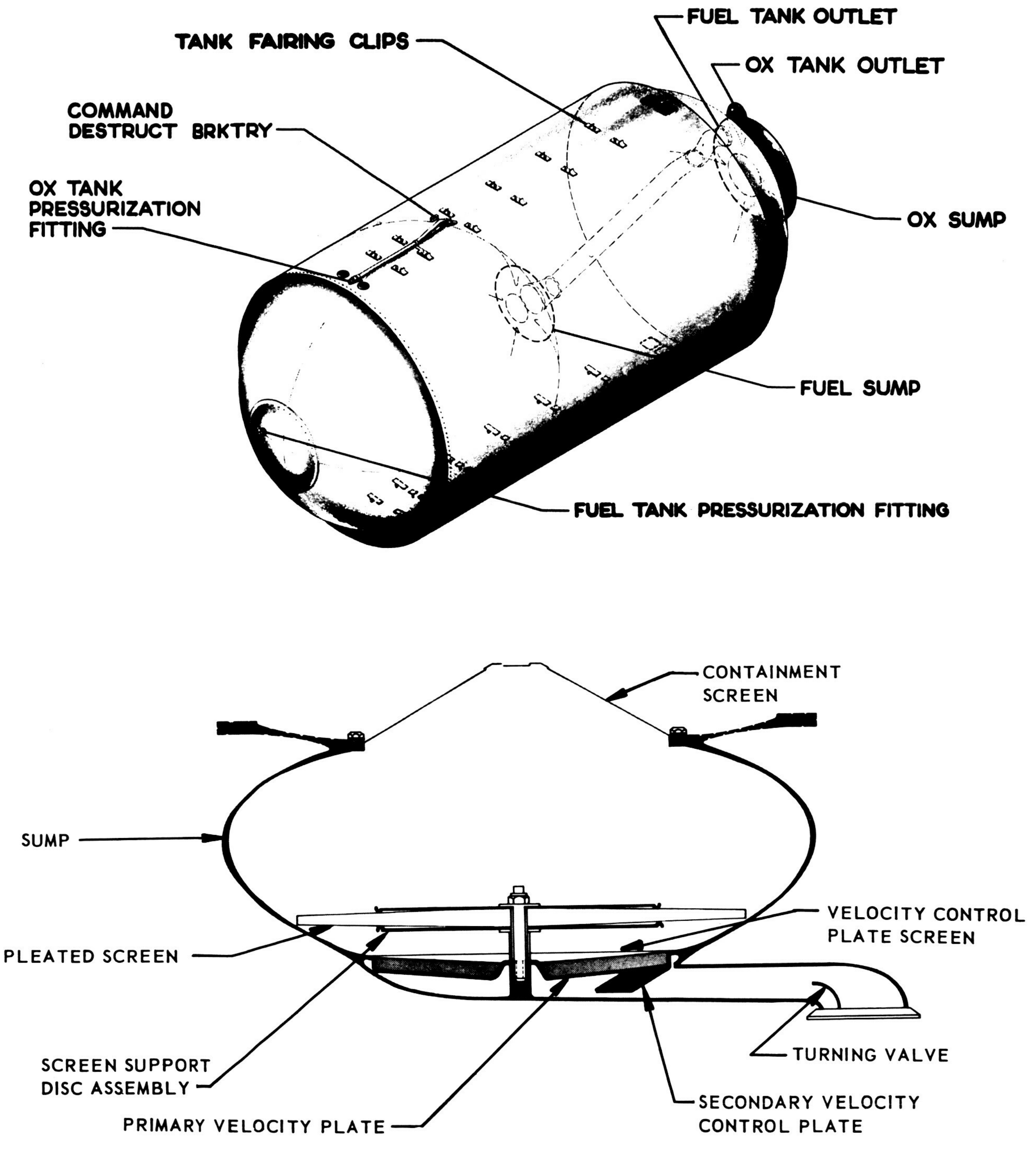

Figure 13-6 Tank Section and Propellant Sump Detail

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13.2.3 Aft Section

The aft section provides structural continuity from the tank section to the booster adapter and supports the rocket engine. The aft rack structure is comprised of magnesium structural members, magnesium skins, and tubular aluminum braces. The aft equipment rack provides mounting facilities for gas storage, basic optional, and other equipment items. Certain mission-oriented equipment items (i.e., secondary propulsion system, additional gas, solar arrays, etc.) may also be accommodated.

The aft section extends 79.5 inches from Station 383.84 to Station 462.5 and terminates in an aft bulkhead which supports the attitude control jets and booster separation guide rollers. The guide rollers engage the four guide rails mounted on the inside of the booster adapter. The aft section is shown in Fig. 13-7.

13.2.4 Booster Adapter

The booster adapter, shown in Fig. 13-8, provides the structural interface between the Standard Agena and the first-stage booster. The basic adapter is designed to mate with the TAT booster. An optional 34-inch skirt extension is used to accommodate the larger diameter of the Atlas booster. The adapter length, including both the basic adapter and the Atlas extension, is 142 inches.

When attached to the Standard Agena, the adapter encloses the aft section and extends from Station 384 to Station 526 (Atlas). After final mating with the booster and Standard Agena, the adapter remains permanently attached to the booster. The booster adapter contains two retrorockets mounted at the 0- and 180-degree angular positions. The separation components consist of four separation roller rails and a mild detonating fuse (MDF) separation device which is installed on the interior of the adapter. The adapter also houses a self-destruct system that is designed to destroy the vehicle in the ascent phase in the event of a hazardous or uncontrollable flight condition.

BASIC VEHICLE

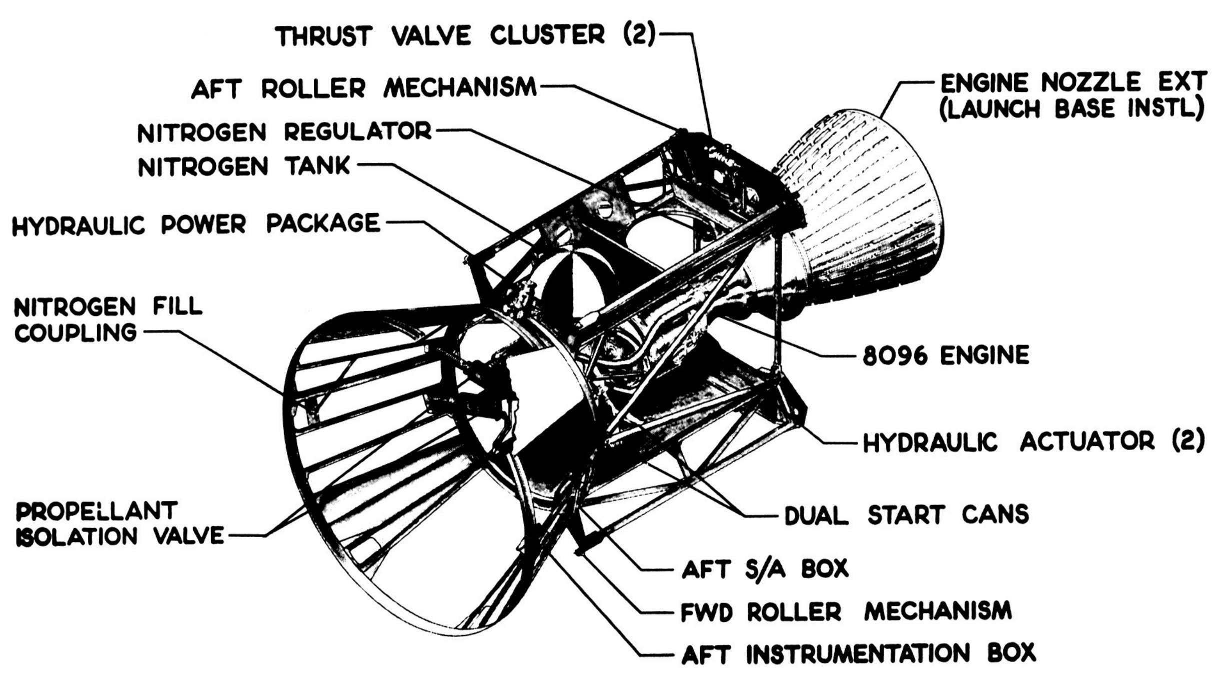

Figure 13-7 Aft Section, Basic Vehicle

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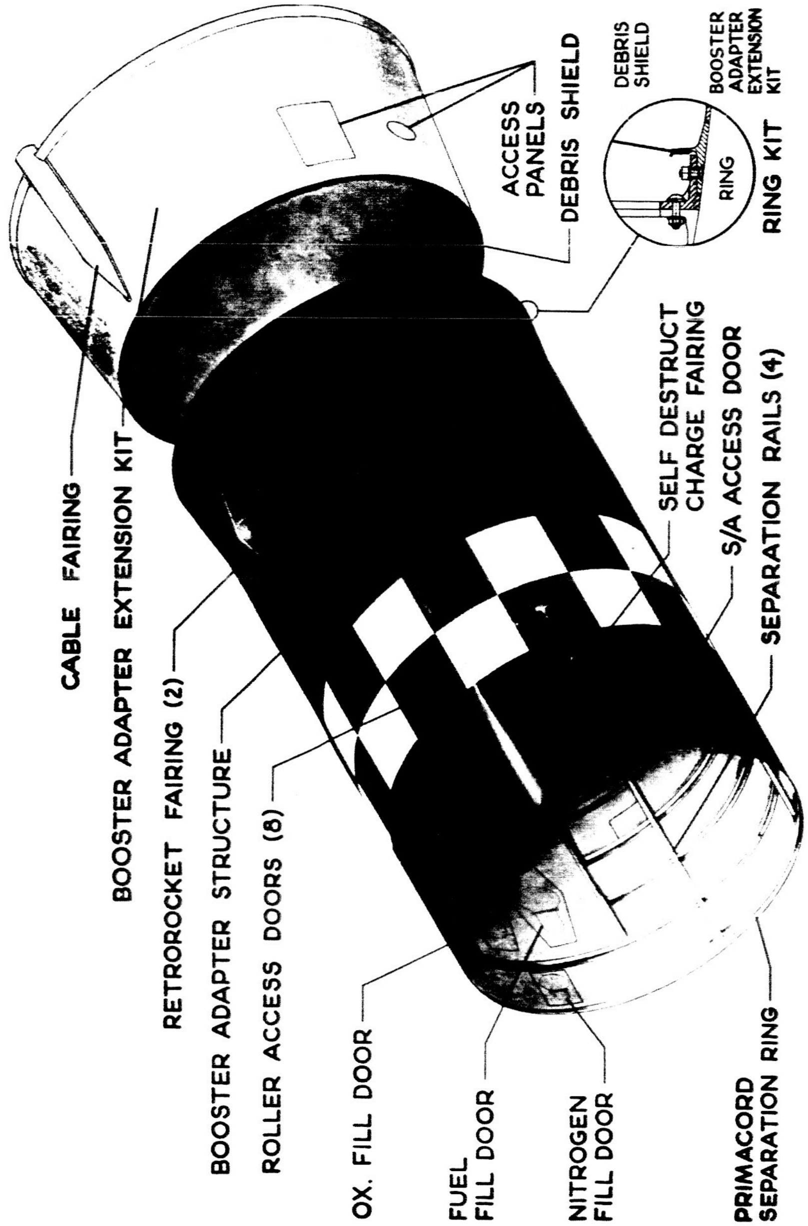

Figure 13-8 Booster Adapter, Basic Vehicle

An alternate destruct system, command destruct, may be installed in the Agena to replace the self-destruct system. The two systems are discussed in detail in Section 15. Provision is made for access to components of the aft section and for interface with launch pad AGE.

13.3 OPTIONAL SPACEFRAME HARDWARE

Optional spaceframe hardware available for use on prospective spacecraft missions is primarily limited to equipment that provides for the removal and/or installation of other subsystem equipment. This equipment includes an aft rack structural kit, adapter kits, and the booster adapter extension and extension ring kits.

13.3.1 Aft Rack Structural Kits

Program equipment may be installed in the Agena aft rack in some instances. Optional kits, right and left aft structural, are available to strengthen the aft equipment rack to accommodate the additional weight of aft rack mission equipment or payloads of up to 1,000 lb. These kits are designed and constructed in accordance with LMSC specification 1414908. The available space is shown in Fig. 13-9.

13.3.2 Optional Equipment Adapter Kits

Optional equipment adapter kits that provide for the installation of the various optional subsystem equipment are available. Provisions for installing these kits are included in the basic Agena.

13.3.3 Booster Adapter Extension Kits

These items of optional hardware are required when the Agena is combined with the Atlas booster. The booster adapter extension kit (LMSC Specification 1414562) is furnished by the Air Force to GD/A and assembled to the Atlas

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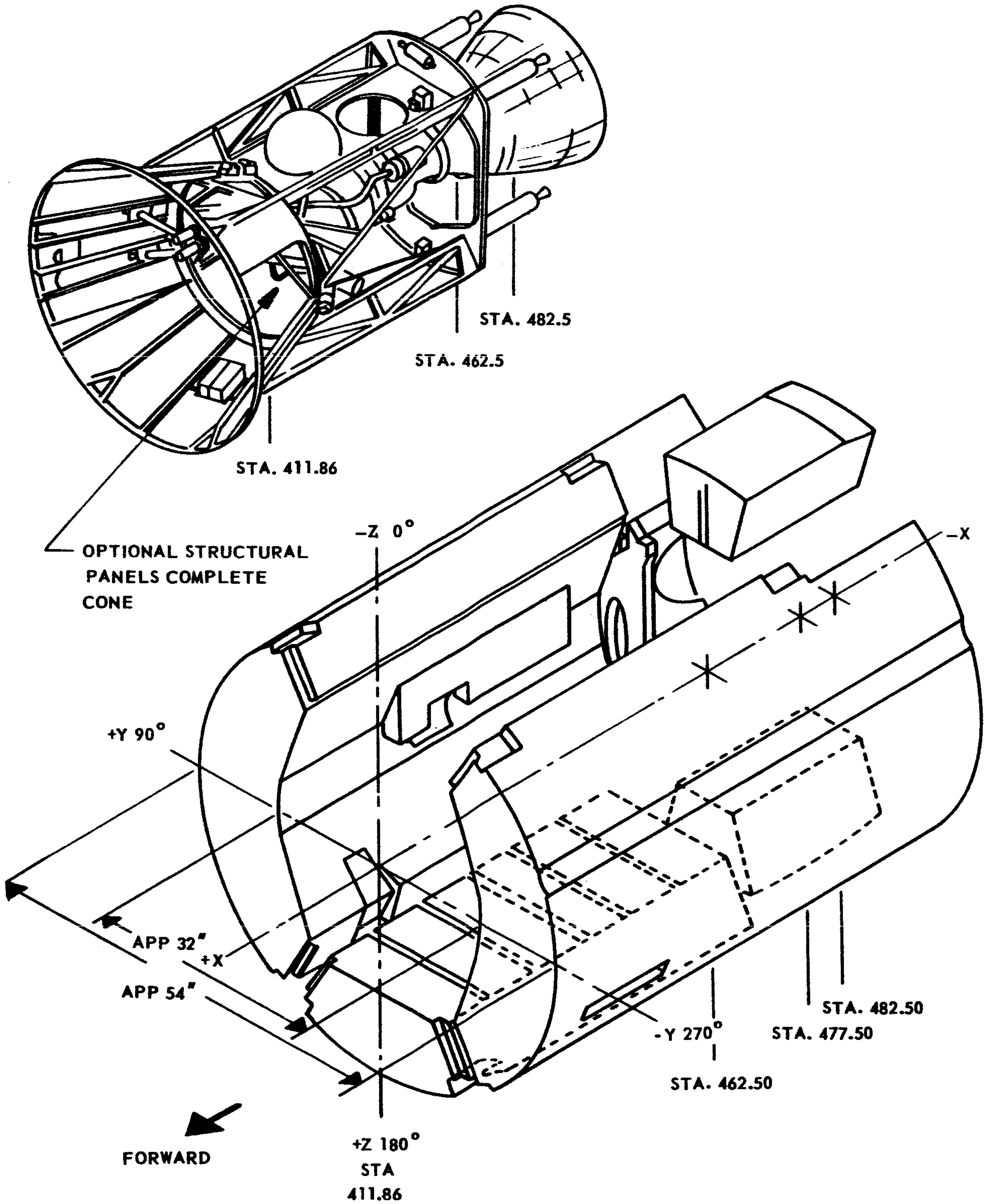


Figure 13-9 Agena D Aft Section - Space Available for Program Usage

booster. The booster adapter extension ring kit (LMSC Specification 1414712) is attached to the booster adapter (discussed in par. 13.2.4) and mates with the adapter extension kit discussed above.

13.4 PROGRAM PECULIAR SPACEFRAME: AUXILIARY RACKS

The peculiar spaceframe hardware that is of significant importance to the spacecraft contractor consists primarily of the shrouds and adapters, discussed in Sections 10 and 11, and auxiliary racks discussed below. The existing Agena forward rack has some space available for the installation of spacecraft support equipment. Additional space may be provided by adding an auxiliary forward equipment rack to the Agena. Four auxiliary forward rack designs are available, one developed for the Gemini Program and three for the 698BK program. Configuration details of the auxiliary racks are shown in Figs. 13-10 and 13-11.

E-3236-5

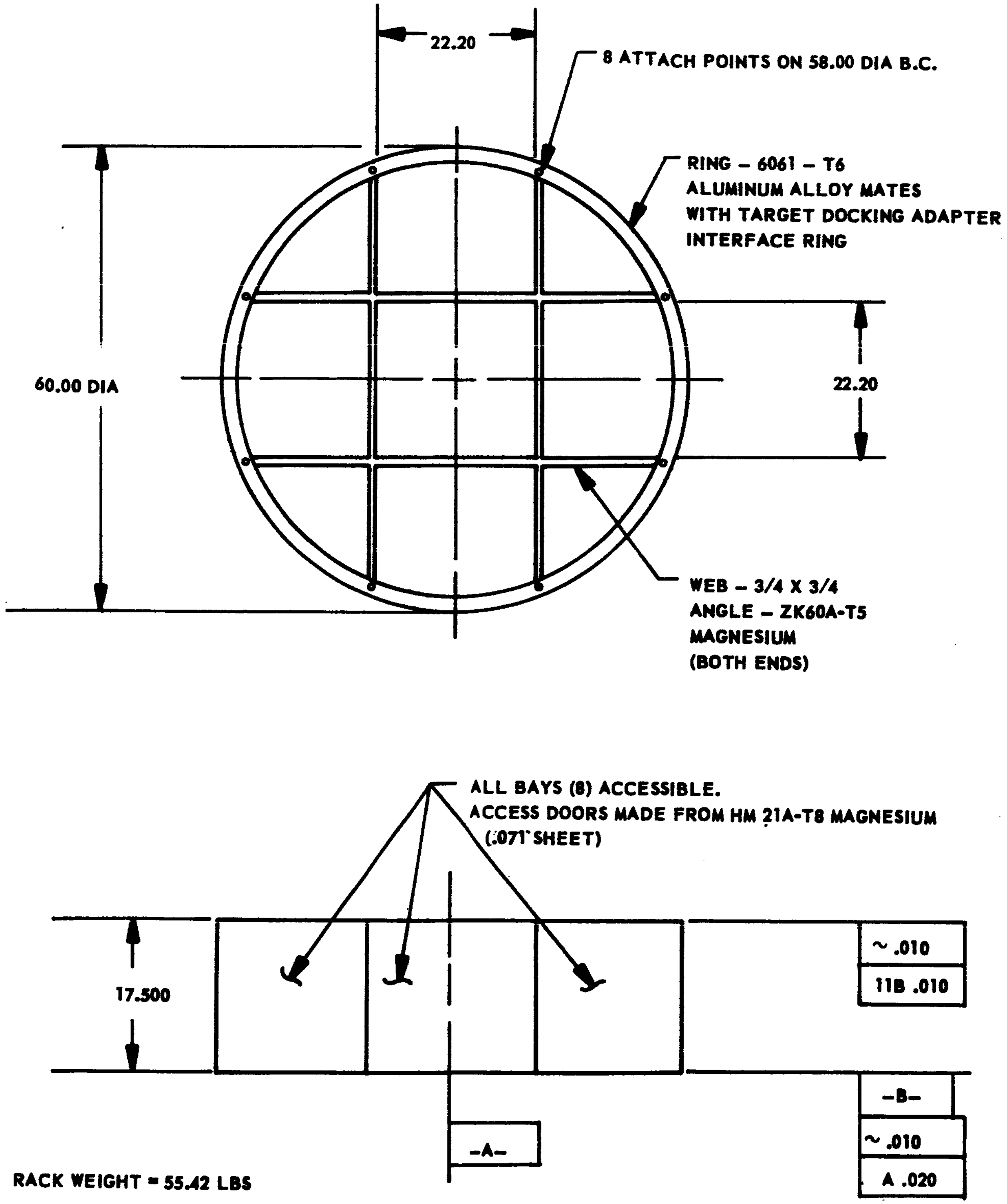
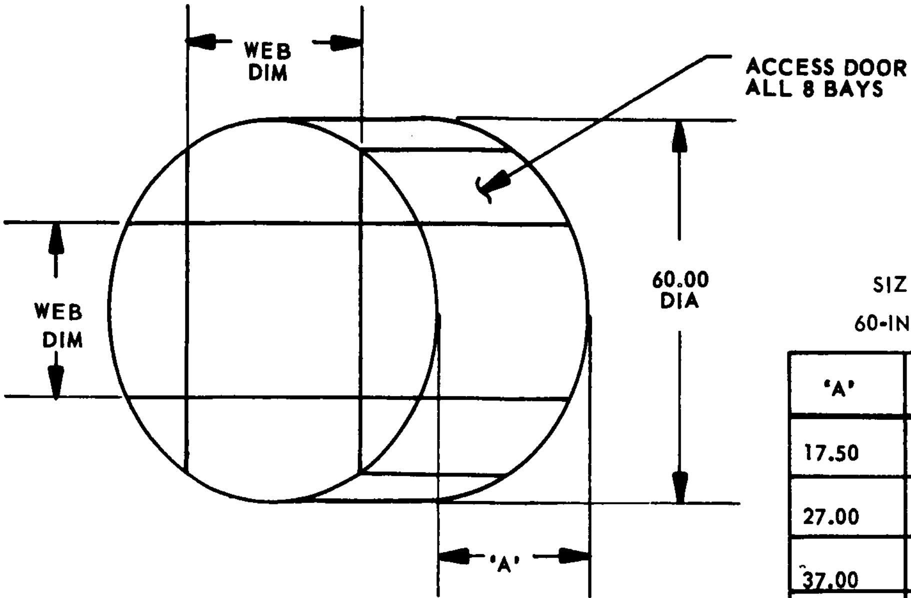


Figure 13-10 Auxiliary Rack, Gemini ATV



SIZES OF AVAILABLE
60-IN. CONFIGURATIONS

'A'	WEB DIM	WEIGHT (LB)
17.50	22.20	55.4
27.00	22.20	106.0
37.00	22.20	127.0
34.00	41.00	84.0

72-IN. HAMMERHEAD CONFIGURATION

ACCESS – FWD AND AFT ENDS ONLY

WEIGHT – 112 LBS

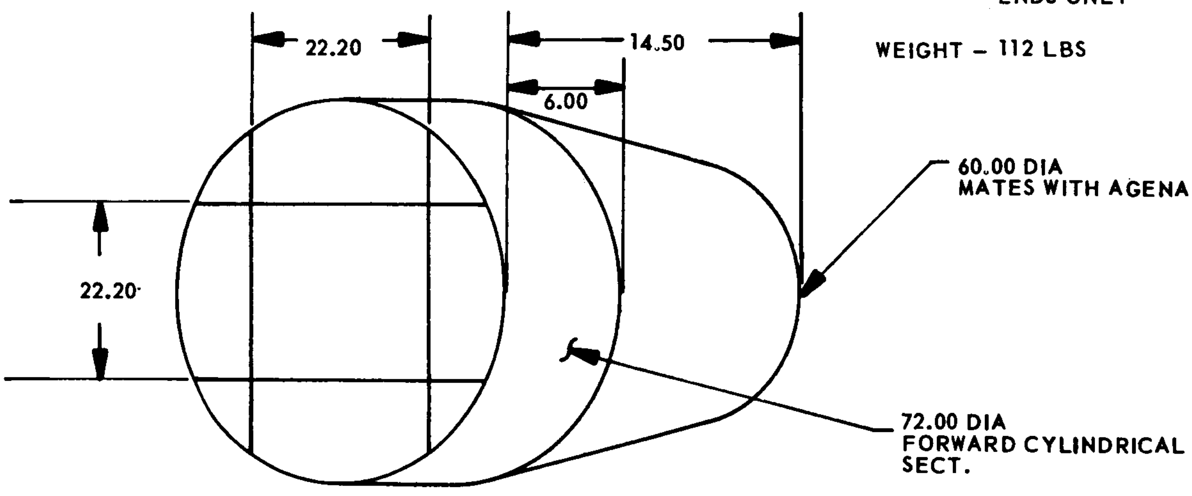


Figure 13-11 Auxiliary Racks

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SECTION 14
PROPULSION SUBSYSTEM

14.1 GENERAL

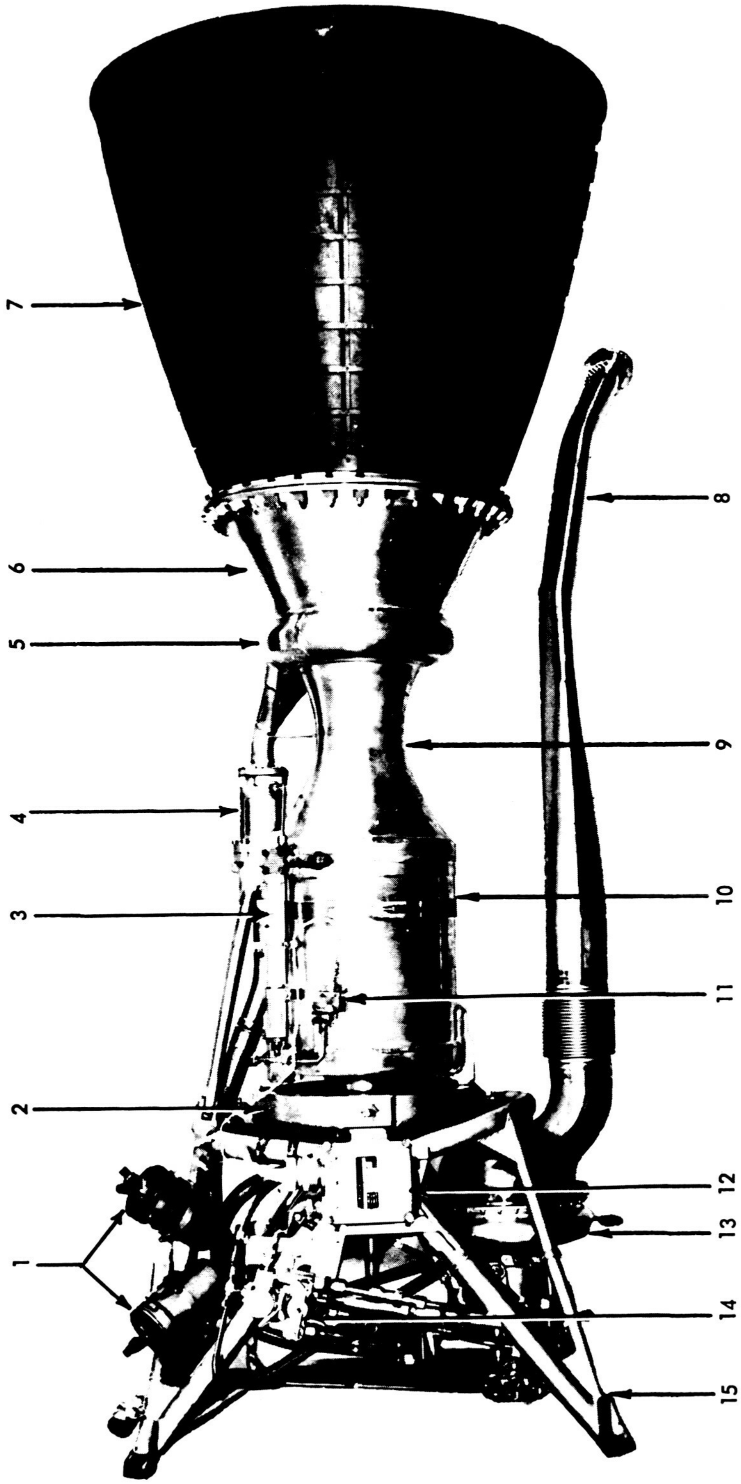
The propulsion subsystem includes engines, motors, and pyrotechnic devices used by the Agena in the performance of its mission as an intermediate stage booster and/or as an orbital vehicle. The principal unit in the propulsion subsystem is the main or Model 8096 primary propulsion system (PPS), which is capable of performing single- or dual-burn missions.

For missions requiring a greater number of restarts, an optional multistart PPS (Model BAC 8247) engine, which has been developed for the Gemini/Agena program, can be used. Other optional equipment developed and qualified by the S-01B/SS-01B program is described under par. 14.3.

Qualified program peculiar hardware is described in par. 14.4. This includes the secondary propulsion system (SPS), solid propellant rockets and pyrotechnic actuator devices.

14.2 BASIC PROPULSION SYSTEM

The propulsion system consists principally of the BAC Model 8096 (USAF YLR-81-BA-11) rocket engine (Fig. 14-1). Agena propulsion may be considered in terms of its three component systems: propellant tank pressurization; the engine; and the feed, load, and vent system. Figure 14-2 shows the propulsion system schematic for the basic 8096 engine for both the single and dual-start configurations. (Also shown are the peculiar features of the multistart Model 8247 engine.)



1. TURBINE STARTER ASSEMBLIES
2. ENGINE GIMBAL RING
3. NITROGEN FAST-SHUTDOWN TANK
4. OXIDIZER VALVE
5. OXIDIZER MANIFOLD RING
6. DIVERGENT NOZZLE SECTION
7. THRUST CHAMBER NOZZLE EXTENSION
8. TURBINE EXHAUST DUCT
9. NOZZLE THROAT SECTION
10. THRUST CHAMBER
11. THRUST CHAMBER PRESSURE SWITCH
12. ENGINE RELAY BOX
13. TURBINE
14. GAS GENERATOR BIPROPELLANT VALVE
15. ENGINE MOUNT STRUCTURE

Figure 14-1 BAC Model 8096 Agena Engine

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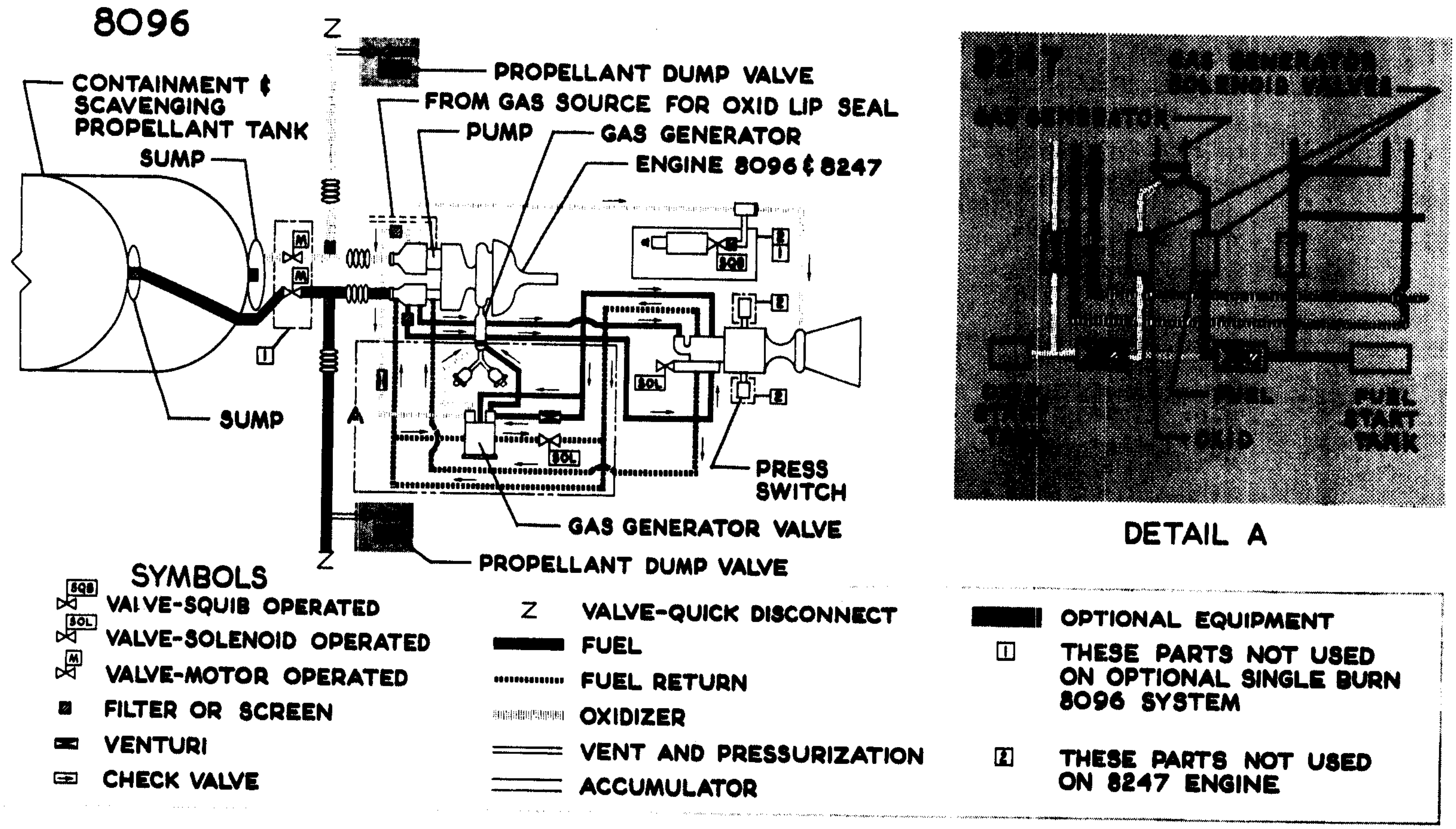

Figure 14-2 Propulsion System Schematic for Model 8096 and 8247 Engines

14.2.1 Rocket Engine System, Model 8096

The basic Model 8096 engine system is a dual-start, liquid bi-propellant type with a single combustion chamber. The engine develops a rated thrust of 16,000 pounds in a vacuum for a total duration of 240 seconds. Fuel is unsymmetrical dimethylhydrazine (UDMH); oxidizer is inhibited red-fuming nitric acid (IRFNA). (See par. 14.3 for single start.)

The thrust chamber assembly is an integral unit consisting of the combustion chamber, nozzle throat section, divergent nozzle section, and a radiation cooled titanium nozzle extension. The expansion ratio is 45:1. The thrust chamber and nozzle are regeneratively cooled by oxidizer entering through passages in the thrust chamber wall before injection into the combustion chamber. The fuel and oxidizer ignite hypergolically in the rocket engine thrust chamber.

An engine electrical control system provides for starting, operating, and shutting down the rocket engine in response to signals from the Agena guidance and control system. The engine is gimballed and provides vehicle attitude control in pitch and yaw during engine operation by means of lateral and vertical thrust chamber movement.

The turbopump assembly supplies fuel and oxidizer under pressure from the propellant tanks to the rocket engine thrust chamber and gas generator. The assembly consists of a single-stage-impulse type turbine; a fuel pump and an oxidizer pump, which are gear-coupled to the turbine shaft; and a gear housing that serves as the assembly frame.

The gas generator provides the hot gases required to start and maintain the turbine operation which, in turn, drives the propellant pump. The gas generator assembly consists of a small combustion chamber, single engine starter-igniter, a gas generator bi-propellant valve and its associated solenoid valve, and the gas generator fuel and oxidizer venturis.

Control of the propellant flow from the turbopump to the engine thrust chamber is provided by a pressure-operated oxidizer valve and by a fuel valve.

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14.2.2 Pressurization System

The pressurization system uses high pressure helium gas to pressurize the propellant tanks to provide the required propellant pressures at the rocket **engine propellant pump inlets. The system (Fig. 14-3) consists of gas storage, pyro-operated helium control valve, and the associated tubing that connects** these components to the propellant tanks and the fill and dump couplings.

The pressurization system maintains the desired pressures in the propellant tanks at the engine turbopump inlets. It maintains these pressures from the start of the countdown (propellant loading operation) until the completion of rocket engine shutdown. Approximately 1.5 seconds after start of the initial engine burn, the pyro-operated helium control valve (POHCV) opens in response to a signal from the guidance and control system sequence timer. When the control valve is open, helium gas flows through fixed orifices (one for oxidizer and one for fuel) to pressurize the propellant tanks. After a predetermined interval of time, the oxidizer tank is isolated by closing a portion of the POHCV. The isolation of the oxidizer tank serves two purposes: (1) creates a sufficient bias to maintain the internal pressure of the fuel tank above that of the oxidizer tank and (2) prevents oxidizer vapors from mixing with fuel vapors through the pressurization system plumbing. This pressurization of the propellant tanks is sufficient for subsequent engine operations.

14.2.3 Propellant Feed, Load, and Vent System

The feed, load, and vent system is comprised of the following main components: two propellant-fill couplings, two propellant-vent couplings, one helium-fill coupling, four strainers, four bellows, and two ullage-control tubes which are a part of the propellant tank assembly. The system interfaces with launch pad AGE to provide for loading propellant and helium port

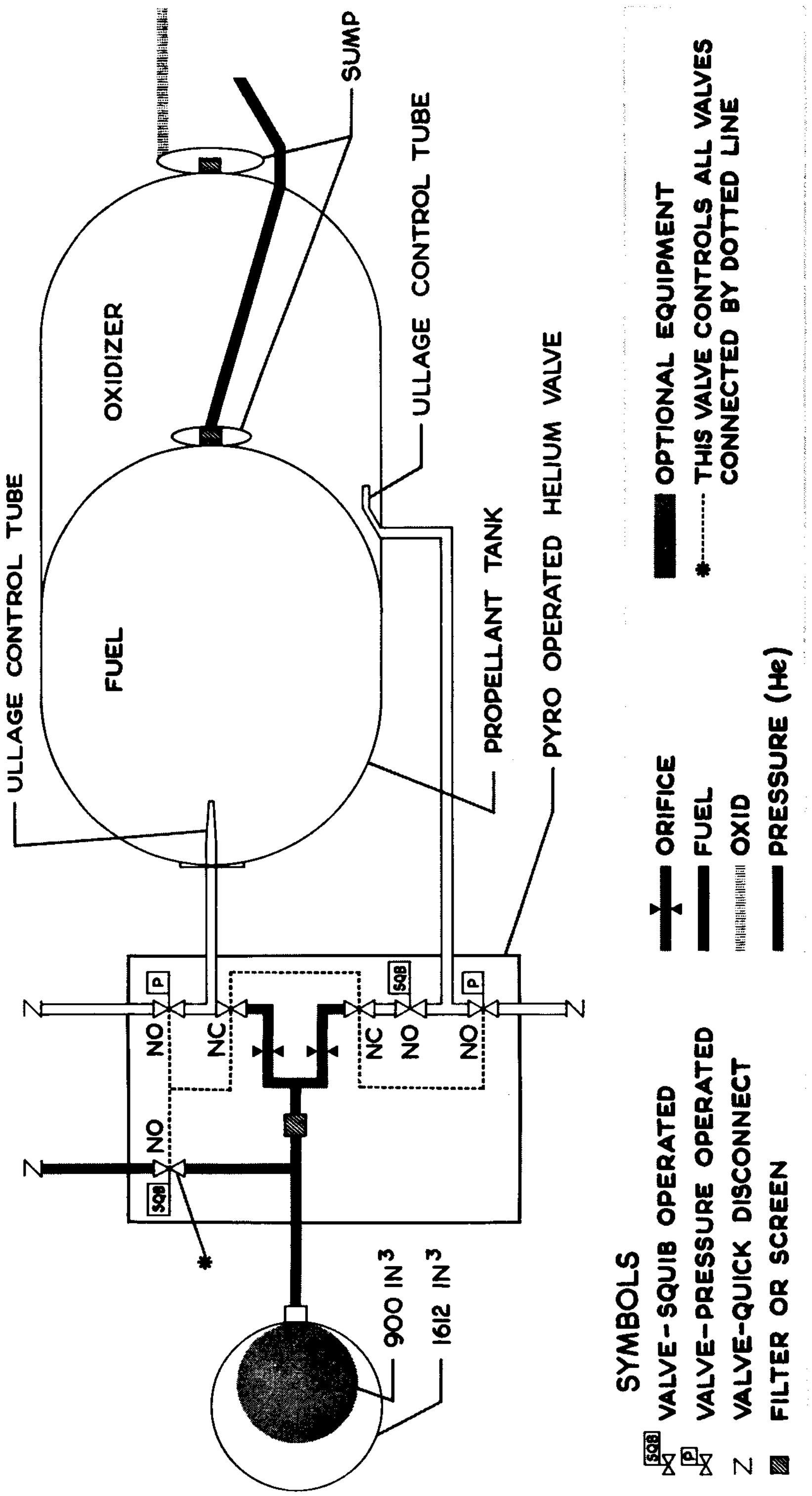

Figure 14-3 Tank Pressurization System

closure after vehicle liftoff to prevent leakage. The system also permits dumping of the propellants out of the tanks and helium from the helium spheres.

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14.3 OPTIONAL PROPULSION HARDWARE

The optional kits described in this section are: the USAF Model XLR-81-BA-13 rocket engine kit; the single start kit; and the propellant dump kit. These items are qualified for flight and may be installed on the basic Agena to provide additional propulsion system capability.

14.3.1 Rocket Engine System, Model 8247

Missions that require multicycle engine capability (three or more) may utilize the BAC Model 8247 rocket engine kit (LMSC Specification 1414829). The kit consists essentially of a multi-start rocket engine, two start tank pressure transducers, wire harnesses, safe and arm plugs, tubes, and associated attachment hardware.

The multi-start engine consists of a thrust assembly, turbine pump assembly, an overspeed shutdown electronic gate and cable assembly, and additional engine components. The engine is basically the Model 8096 engine with the solid propellant start charges replaced by start tanks (Fig. 14-4) containing hypergolic UDMH and IRFNA. The engine is started by allowing start tank propellants to flow to the gas generator where hypergolic reaction takes place. The generated gas drives the turbine which in turn drives the oxidizer pump and fuel pump. These pumps supply the thrust chamber and gas generator with propellants; and, after each start, recharge the start tanks. The duration of initial burn and subsequent restart firings (up to 4) may be of any desired length, providing that no engine burn duration is less than two seconds. The total engine burn time, however, may not exceed 240 seconds. Initial testing has been done on extending the number of restarts to fourteen. Total thrust of the engine, including exhaust duct thrust, in vacuum environment is 16,000 pounds.

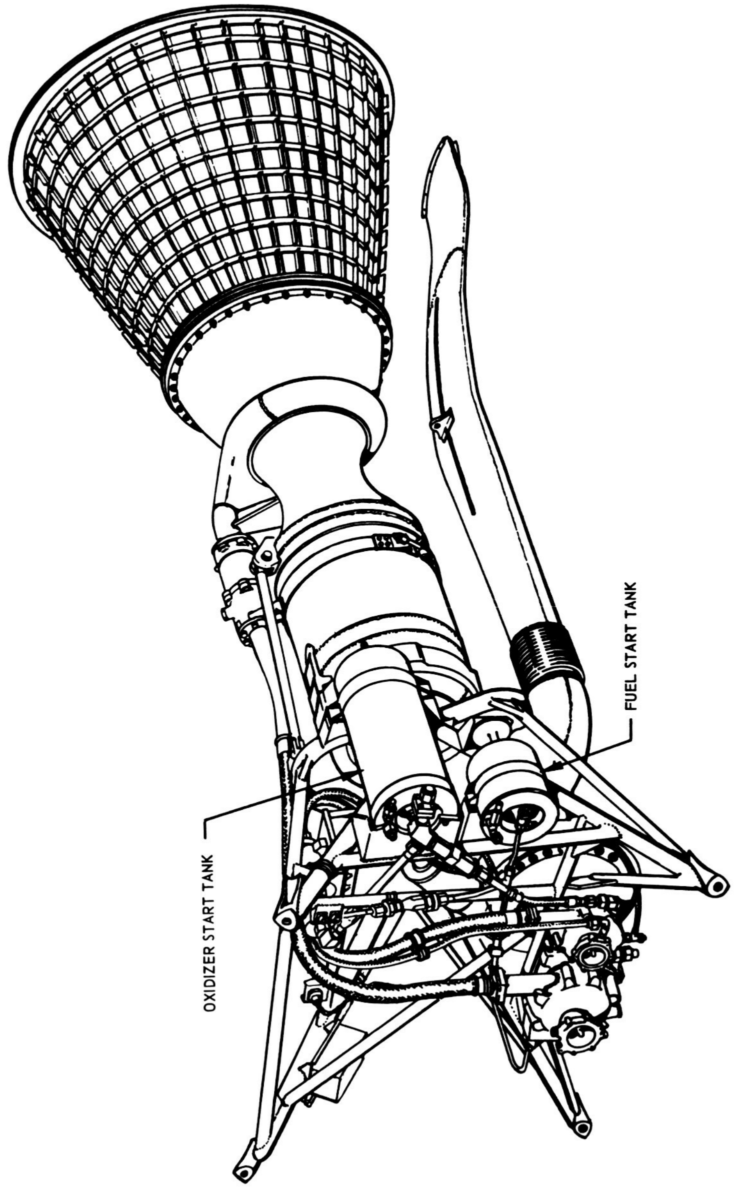

Figure 14-4 Optional Model 8247 Multi-Start Engine

The turbine overspeed system consists of a magnetically sensitive motion pickup, electronic network circuit, and relays. Function of the system is to terminate the flow of propellants to the gas generator and thrust chamber if the turbine speed should exceed 29,500 ± 500 rpm.

14.3.2 Single Start Kit For Model 8096

Missions that do not require the additional performance afforded by the Model 8096 dual burn capability may utilize the single start kit. This kit replaces the basic dual start components. Single start results in a higher Agena vehicle reliability, but results in a loss in payload weight capability for certain missions. LMSC Specification 1414830B applies to this kit.

14.3.3 Propellant Dump Kit

Programs which require the Agena to be actively attitude stabilized or to maintain passive attitude control over a long period of time (hours or days) normally use this kit (See Section 16 for details of these modes). Dumping the residual propellants prevents satellite attitude disturbance that could result from leakage of residual propellants during orbit.

The propellant dump kit (LMSC Specification 141831) consists of two pyrotechnically actuated valves, tubing,and electrical wiring harness. The installed kit provides a means for rapidly dumping residual propellants overboard while on orbit. The rate at which the propellants dump is a function of the tank pressure, tank temperature, and the amount of residuals remaining at the end of engine burn. The propellants are dumped through nullifier fittings in order to minimize vehicle torque.

14.4 MISSION PECULIAR HARDWARE

Major items of mission peculiar propulsion equipment available for use with the basic Agena are the Gemini program peculiar secondary propulsion system

(SPS), impulse rockets and an assortment of program peculiar pyrotechnic devices. Unlike the optional kits, some applications of the listed peculiar hardware may require vehicle modification to provide for installation.

14.4.1 Secondary Propulsion System (SPS)

A secondary propulsion system has been developed for use on the Gemini program that is available for use on other programs as an item of program peculiar equipment. The system is a completely contained unit (module) including positive expulsion propellant tanks, a gas pressurization system, one 16-lb thrust chamber assembly (Unit I), and one 200-lb thrust chamber assembly (Unit II). The SPS modules illustrated in Figs. 14-5 and 14-6 can be installed in pairs or singly on the Agena D aft rack. The thrust chamber combinations may be varied to include either two Unit I or two Unit II thrust chambers. The complete module can also be modified to function as a pressurization and propellant storage module for use with remotely located thrust chambers. Leading particulars of the SPS are listed in Table 14-1.

Table 14-1
SPS LEADING PARTICULARS

	Unit I	Unit II
Thrust (Vacuum)	16 lb	200 lb
Max. No. cycles	90	20
Cycle duration	0.5 to 150 sec	0.5 to 50 sec
Module Weight		
Wet	303.8 lb	
Dry	126.4 lb	
Total system impulse	40,000 lb sec	
Operating temp. range	0° to $+100^{\circ}$F	

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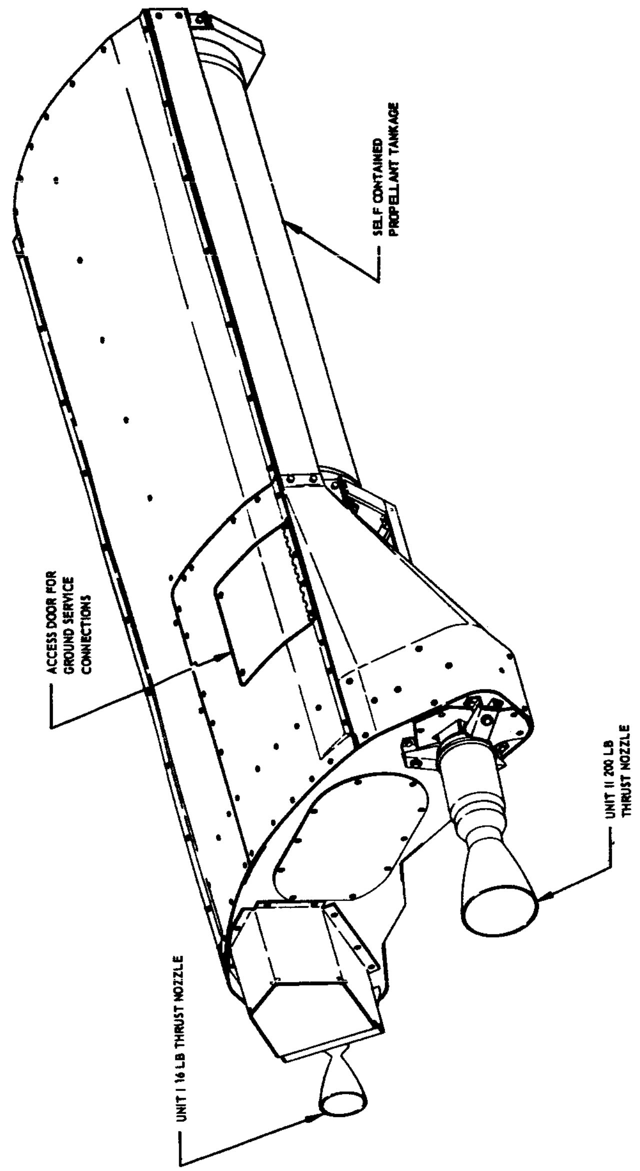

Figure 14-5 Secondary Propulsion System Module

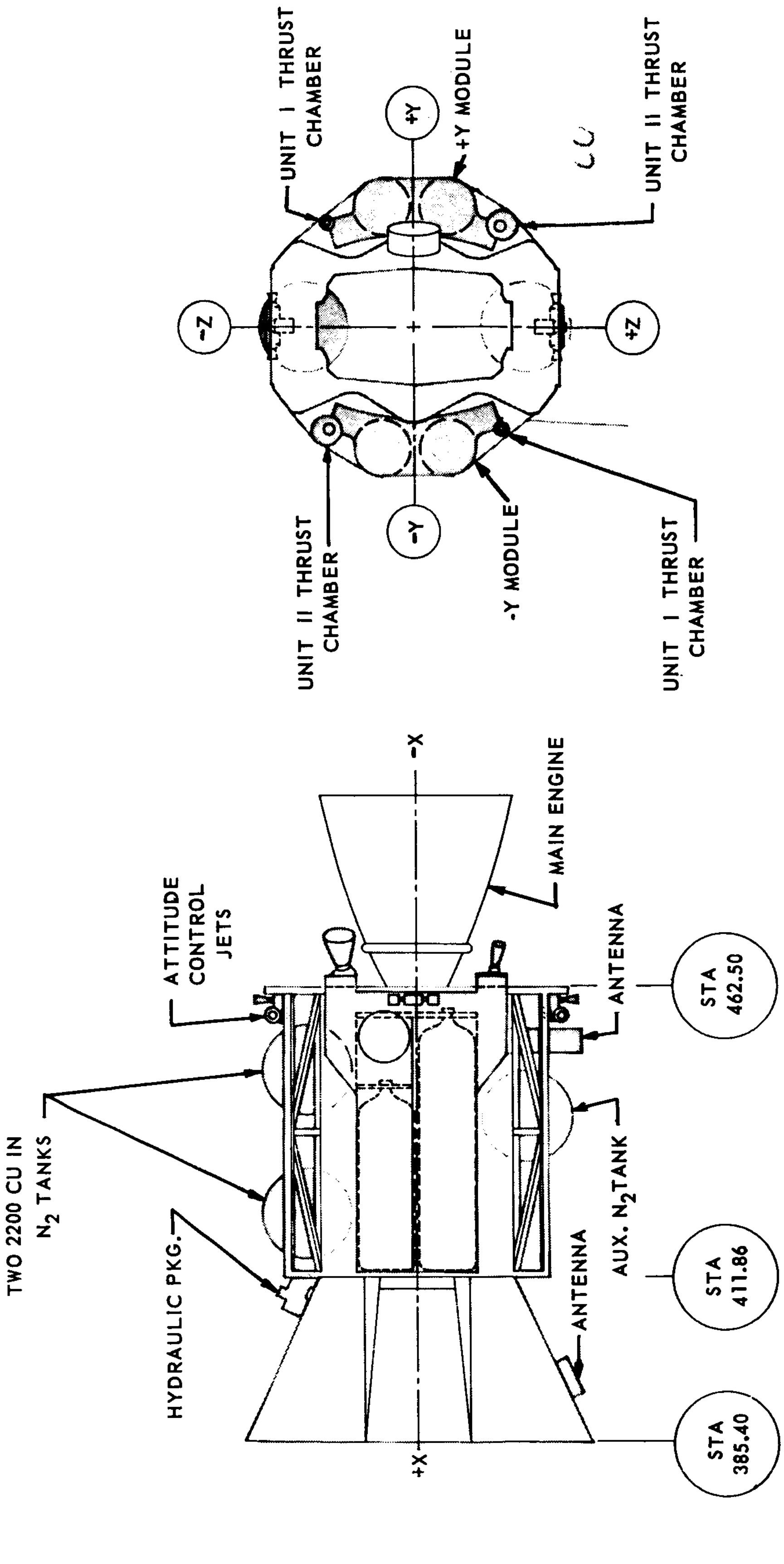

Figure 14-6 Typical SPS Installation on Aft Rack (Gemini)

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14.4.2 Fixed Impulse Rockets

Three flight-qualified, fixed-impulse rockets are available as program peculiar equipment. The Type III retrorocket (Fig. 14-7) has been mounted on the aft rack of the probe Agenas to provide the thrust necessary to perform a retro-maneuver to avoid Agena impact on the moon or planets. The other retro-rocket is used to provide booster retro thrust during Agena/booster separation. The spin rockets were used on the S-27 spintable to provide the necessary thrust for spintable spin-up. The above items could be used to provide a fixed impulse for other functions if required.

Retro and spin rocket data is listed in Table 14-2.

Table 14-2
FIXED IMPULSE ROCKETS

Parameter	Agena Retrorocket Type III (Hercules Powder Co.)	Booster Retro-Rocket (Rocket Power Inc.)	Spin Rocket 1KS30HA (DAC)
Weight	15 lb	4.69 lb	0.75 lb
Length	20 in	15.25 in	5.25 in
Diameter	4.5 in	2.9 in	1.5 in
Propellant	Solid-end burning	Solid-Internal Burning	Solid-Internal/External Burning
Burn Time	15.9 ± 1.6 sec	0.925 sec	0.75 sec
Thrust (avg)	137 ± 23 lb	490 lb	38 lb
Total Impulse	2223 ± 99 lb-sec	455 lb-sec	30 lb-sec

Table 14-3

PECULIAR PYROTECHNICS

Pyrotechnic Device	Design Data	Typical Application
Pin-puller LMSC-1318757	Pin dia: 0.46-in, pin extension: 1.64-in, pin retraction: 1.65-in, nominal preload: 3,800 lb (shear)	1) Recovery chute deployment. 2) H/S torque bar positioning.
Pin-pusher LMSC-1310933	Pin dia: 0.25 in, pin extension: 1.00-in, nominal preload: 3,600 lb (shear)	Fairing Jettison
Explosive bolt LMSC-1354376	Dia: 11/16-in, length: 3.19-in. nominal preload: 6,800 lb (tension)	Shroud V-band clamp release
STL release assembly (double-ended) LMSC-1618709	Bolt dia: 0.25-in, length: 4.26-in, nominal preload: 2,600 lb (tension)	Spacecraft V-band clamp release
Spin-off disconnect (electrical) LMSC-1347315	55 pins or sockets available	Spacecraft/Agena electrical interface disconnect
Squib-operated valves LMSC-1398622	0.25 or 0.50-in line size; working pressure: up to 3,600 psig; may be operated normally closed or normally open	Open and close helium supply in Agena

14.4.3 Peculiar Pyrotechnics

A variety of qualified pyrotechnic devices are available as program peculiar equipment. These devices may be utilized to perform functions such as V-band separation, fairing jettison, electrical disconnect, and gas flow control. Qualified pyrotechnic devices are too numerous to itemize. Table 14-3 lists representative types of different devices and their typical applications. Figures 14-8 and 14-9 illustrate two separation devices commonly used in Agena applications. The double-ended, double bridgewire explosive bolt is qualified, but requires fragmentation shielding. The explosive action in the STL separation device is self-contained.

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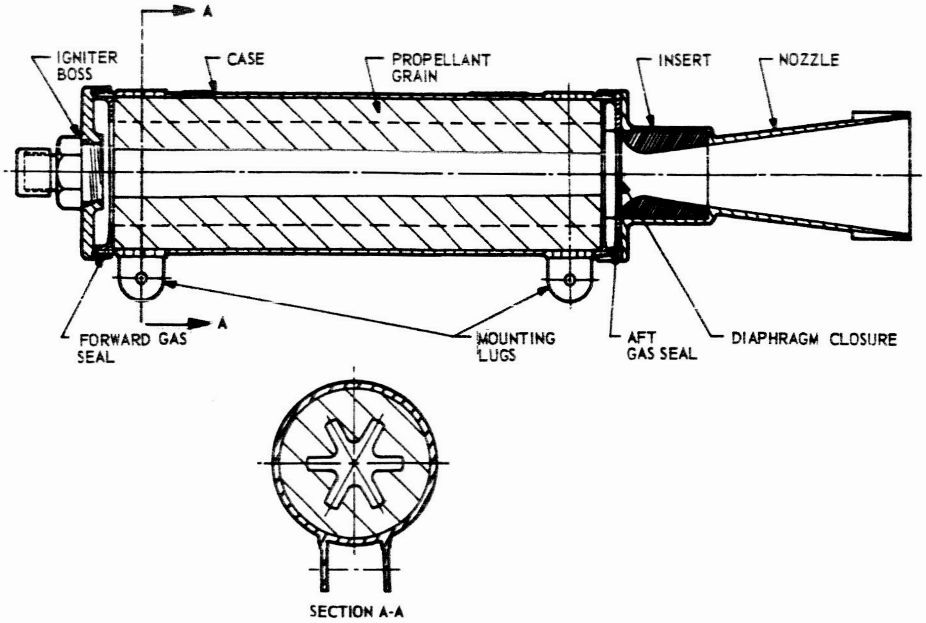

Figure 14-7 Typical Fixed-Impulse Rocket In Agena Use (Type III)

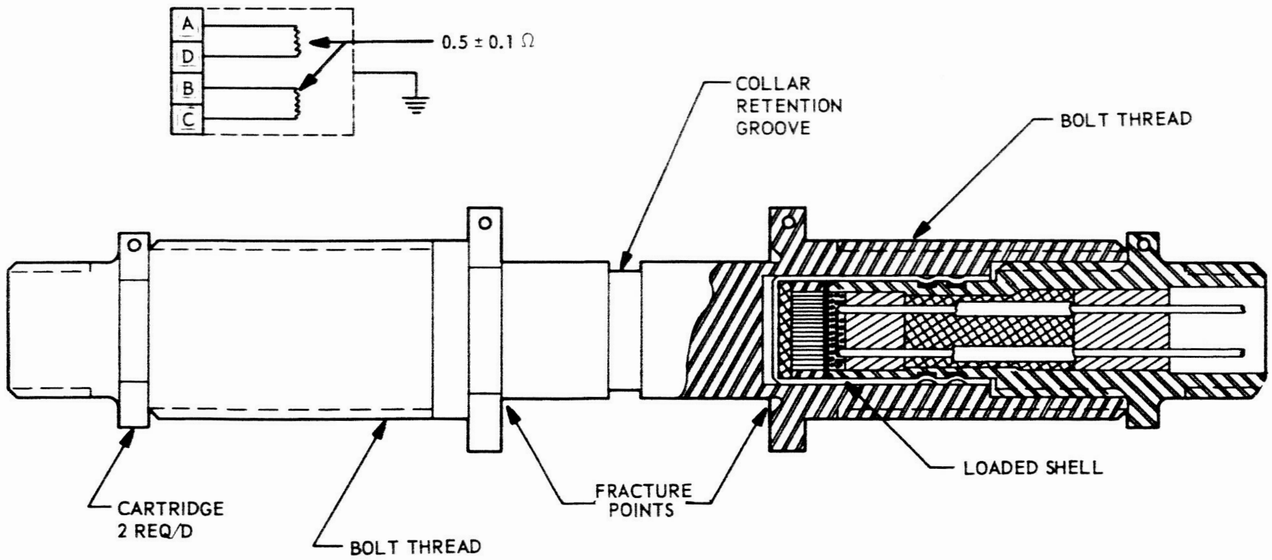

Figure 14-8 Typical Double-Ended Explosive Bolt In Agena Use (McCormick-Selph)

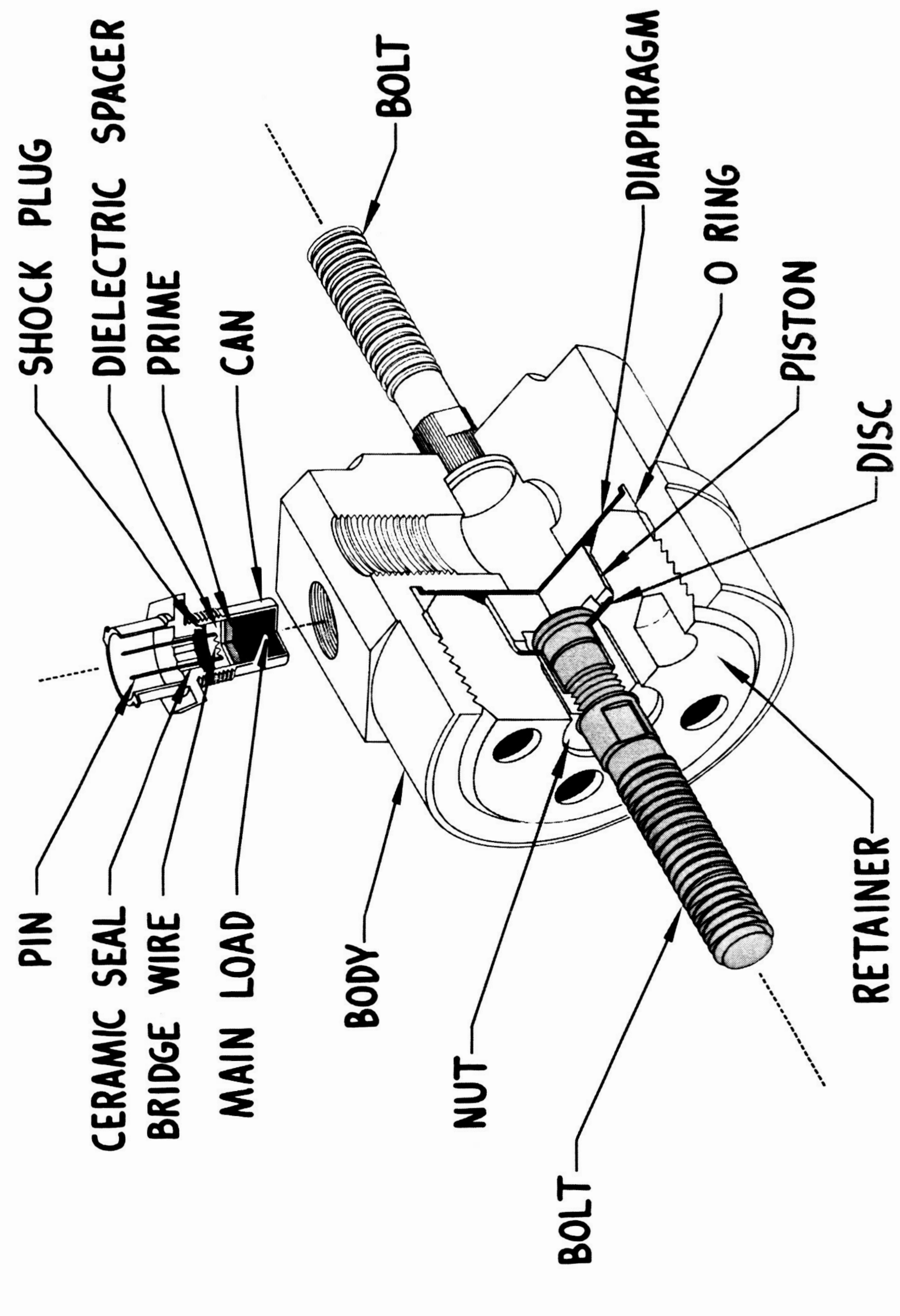

Figure 14-9 Pyrotechnic Release Assembly (STL)

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SECTION 15
ELECTRICAL SUBSYSTEM

15.1 GENERAL

The electrical subsystem comprises two major categories of equipment — one providing auxiliary power and its distribution and the other providing destruct capability. Qualified optional and peculiar equipment is available within these categories to supplement the basic Agena capability in its application to the particular mission.

15.1.1 Electrical Power System

The electrical power system (EPS) furnishes power at the voltage levels and frequencies required by the associated vehicle subsystems and using program equipment for a time period consistent with the vehicle mission duration. The Agena power system is made up of power source components, power conversion components, and power control and distribution components. The power source components consist of primary batteries to supply the initial source of energy to the power equipment and other system components, as well as secondary batteries to supply power to the destruct system.

The power conversion components consist of the 400-cps, 115v ac, three-phase inverter (one of the phases is used for single-phase, 400-cps power) and the ±28v dc-dc converter. Internal vehicle power is controlled by the main power transfer switch which is capable of transferring from AGE power to internal vehicle power. Power control and distribution components consist of switches to transfer between power units, command or programmed switches, and wiring harnesses for distribution of electrical energy to the system components.

15.1.2 Destruct System

The basic Agena may be destroyed by means of the self-destruct system during the ascent phase, liftoff to just prior to booster/Agena separation, in the event that hazardous flight conditions occur. However, the Eastern Test Range (ETR) Range Safety Office requires that any vehicle launched at ETR must have the capability for destruct initiation up to orbit injection (see AFMTC Regulation 80-7, "Airborne Flight Termination Systems"). This requirement is satisfied by replacing (or supplementing) the self-destruct system with an optional Agena command destruct system.

15.2 BASIC VEHICLE ELECTRICAL HARDWARE

The basic Agena electrical hardware makes up the basic power system and the electrical portion of the destruct system. Capabilities of these systems may be supplemented or altered by the addition of optional and peculiar hardware discussed in paragraphs 15.3 and 15.4.

15.2.1 Basic Vehicle Power System

The main battery system of the Standard Agena provides for a component input unregulated voltage ranging between 22.5 to 29.25v dc. The amount of available electrical power depends upon the type and number of batteries selected by the using program. Each installation in the forward section consists of a minimum of two batteries, one to supply the pyrotechnic requirements and one to supply vehicle main power requirements. The pyrotechnic battery, even though electrically interconnected with the primary battery, is in actuality isolated by a diode placed between the main power distribution junction and the pyrotechnic power distribution junction. The diode permits power to flow from the pyrotechnic battery to the main power system; however, the diode prevents power or transient feedback resulting from pyrotechnic activation from being reflected on the main bus. The power distribution system is shown in Fig. 15-1.

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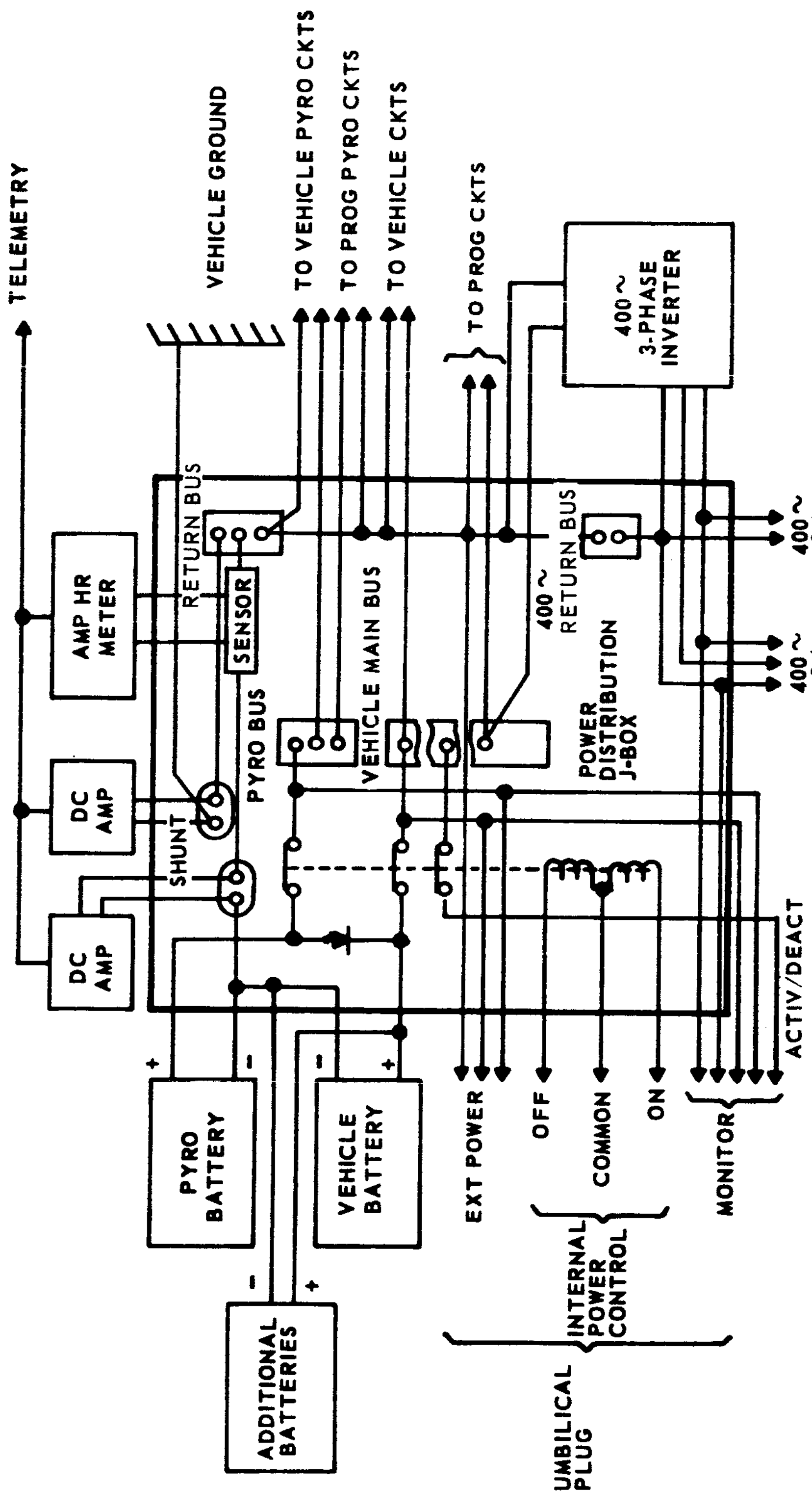

Fig. 15-1 Power Distribution System

The secondary battery system provides power for the destruct systems. Two Type VIA secondary batteries provide the necessary power for the basic self-destruct system. They are installed in the booster adapter assembly. The two Type V secondary batteries used to provide power for the optional command destruct system are installed in the command destruct module.

Table 15-1 lists the nominal battery power required during the period from Agena liftoff through second-burn, earth-orbit injection for a representative one-hour duration ascent mission.

Table 15-1

NOMINAL AGENA POWER REQUIREMENTS

Subsystem	Peak Watts	Average Watts
B (propulsion)	500	6
D (guidance & control)	364	313
C&C (telemetry)	129	116
System losses	108	95
Total	1,101	530

15.2.1.1 Power Source Hardware. Primary batteries furnish the power required by the associated vehicle subsystem for the time periods consistent with the vehicle mission. Secondary batteries furnish the power required for the destruct system. The secondary batteries are rechargeable.

The primary batteries are not part of the basic Agena but must be provided by the using program. Provisions are made, however, for installing a pair of Type IC, Type IVA, or Type VIA primary batteries by utilizing available optional kits. The selection of the type and number of batteries used for each Agena mission is dictated by the power requirements of the using program.

The three sigma capacity rating of the Type VIA battery is 966 watt-hours, and the nominal capacity rating is 1320 watt-hours. (See Table 15-4 for additional battery characteristics.)

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The Type IVA primary battery is a high discharge rate battery and provides the primary source of electrical power for the majority of the NASA Agena vehicles. The 3 sigma capacity rating is 340 watt-hours; the nominal capacity rating is 380 watt-hours. (See Table 15-4 for additional battery characteristics.)

The Type IC primary battery is a medium discharge rate battery. The 3 sigma capacity rating is 10,700 watt-hours. Though the weight of the battery (118 lb) is over 4 times greater than Type VIA and over 7 times greater than the Type IVA, the watt-hour per pound (100 w-h/lb) is twice that of Type VIA and 4 times that of Type IVA (see Table 15-4.) Long orbit life missions, such as the Gemini Agena, normally use the Type IC batteries.

A pair of Type VIA secondary batteries are used in the basic Agena self-destruct system. The battery is designed to deliver 3 amperes at 6 volts for 1 second. The battery is composed of six cells of nickel cadmium elements in a potassium hydroxide electrolyte.

15.2.1.2 Power Conversion Hardware. The Agena power conversion equipment consists of a Type XIIA three-phase inverter and a dc-dc converter. The inverter is mounted in the power module in the Agena forward section and the converter on the guidance module. The inverter and converter are used exclusively for the Agena guidance and control system. Though the inverter output is available at a connector on the power distribution box for program use, care must be exercised to avoid interference with the operation of the Agena Guidance and Control subsystem. Programs requring ±28 v dc regulated power for payload or program peculiar equipment will require an additional power supply.

The inverter provides 115v ac (rms) ±1 percent, 400 cps ±.02 percent, three-phase power with phase AB used as a source of single-phase power. The output is delta-connected with point B of phase AB grounded external to the inverter. The input may range from 22 to 29.3v dc. The load power factor should be maintained between the limits of 0.8 lagging to 0.95 leading on each phase. The maximum load on any single phase is 100 volt-amperes for phase AB and BC and 60 volt-amperes for phase CA.

The Type IX dc-dc converter is designed to provide a ±28.3v dc (±1 percent) output with input voltage ranging from 22 to 29.25v dc. The converter has an output capability of 60 watt (plus) and 20 watt (minus) with the -28.3 volts "tracking" the +28.3 volts within 1 percent, that is, using the plus output as a reference, the +28.3v dc and the -28.3v dc output values will agree within one percent.

15.2.1.3 Power Distribution. The power distribution J-box is the functional center of the electrical power system. The box is the principal distribution point for unregulated dc power, pyrotechnic power, and single- and three-phase ac power.

The signal conditioners required for monitoring the vehicle electrical power system are located within or are attached to the J-box except for the ±28v dc monitor which is in the guidance J-box. The external/internal power transfer switch is located in the power distribution J-box. Vehicle wiring and electrical connectors provide the interconnection between electrical components.

The transfer switch used to switch Agena power from external (AGE) power to internal power is a reversible, motor-actuated switch. Two of the three pole contacts are capable of handling a continuous 60 amperes resistive load and the other 5 amperes. One of the 60-ampere contacts is used to switch pyrotechnic power to the pyro distribution bus, and the other switches the battery power to the main electrical vehicle bus. (See Fig.15-1.)

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15.2.1.4 Destruct Discrete Box. The destruct discrete box consists of relays, fusistors, diodes, and electrical wiring. Four relays (latching type) are part of the self-destruct system. Their primary function is to enable/disable the destruct system. Five relays are used in the booster/Agena separation circuit and one relay provides +28 voltages to actuate the "uncage gyro" relays and "horizon sensor fairing eject" relays. The 28v ground return for the relays energized by booster-furnished power is isolated from the Agena ground return. Two relays are available for program use. Four fusistors with a nominal impedance of 1.8 ohms are used in the booster/Agena separation pyrotechnic circuits. The destruct/discrete box is located in the booster adapter.

15.2.1.5 Pyrotechnic Electrical Hardware. The basic pyrotechnic electrical hardware consists of an internal/external transfer switch (see par. 15.2.1.3), aft and forward safe-arm J-boxes, safe and arm connectors, and pyrotechnic wire harnesses. Except for the booster/Agena separation and horizon sensor fairing eject functions, the pyro signals are programmed by the guidance and control sequence timer. The timer serves as a backup for the separation and ejection functions which are commanded through the booster discretes. (See par. 15.2.2 for description of the destruct system.)

The pyro electrical power is applied to the primary pyro distribution bus (located within the power distribution box) whenever the transfer switch is in the internal power (battery) mode. The pyro +28v dc power is distributed to the sequence timer, forward safe-arm box, aft safe-arm box, and to the destruct/discrete box (see par. 15.2.1.4.)

The forward and aft safe-arm boxes contain relays and fusistors. A safe-arm receptacle is installed on each of the boxes which are mounted on the Agena in a manner that facilitates mating of the safe or arm connectors. The receptacles also provide access to the pyro circuits and are used for checking and verifying the system. The pyro circuits are safed or armed by the safe (LMSC-1341077) or arm (LMSC-1341078) connector assembly, respectively. The assembly consists of a connector and cylindrical can which encloses jumper wires. Three safe-arm connectors are required per Agena, one each on the safe-arm boxes and one in the booster adapter.

15.2.2 Basic Vehicle Destruct System

The basic Agena destruct system is a self-destruct system which provides destruct capability during the ascent phase, liftoff through just prior to booster/Agena separation. It consists of two Type VIA secondary batteries, a junction box, an initiator, two separation switches, and wiring harnesses. These items are installed in the booster adapter.

The system functions by firing a shaped charge that ruptures both the Agena oxidizer and fuel tanks. The resultant mixing of hypergolic propellant destroys the vehicle. The self-destruct system operates from batteries that are independent of the primary batteries of the vehicle.

The operation of the self-destruct system is initiated in one of two ways.

a. The Range Safety Officer initiates a command signal to the booster command destruct system.
b. Initiation is mechanically switched by an untimely (premature) separation of the Agena from the booster vehicle.

15.3 OPTIONAL ELECTRICAL HARDWARE

Optional Agena electrical power system hardware includes the following: primary and secondary batteries, battery adapter kits, current monitoring equipment, safe and arm plug kits, and a command destruct kit used in lieu of the basic self-destruct system. The above equipment may also be modified to program peculiar use as discussed in par. 15.4. Specification numbers are given in pars. 15.3.1.1 through 15.3.1.3.

15.3.1 Power System

Qualified optional components available for use in the Agena electrical power system include primary and secondary batteries, an ampere-hour meter counter and sensor, and safe and arm plug kits.

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15.3.1.1 Power Source. Both primary and secondary batteries are available to furnish power to the Agena and payload. Space provisions are provided on the aft equipment support structure which may be used for the installation of additional batteries by the using program.

The primary batteries contain positive electrodes of zinc and use potassium hydroxide as the electrolyte. They have high-energy ratings and are not rechargeable.

Optional primary batteries include Type IVA, Type VIA, and Type IC. Separate adapter kits required to accommodate their installation are as follows:

Optional Kit	Spec. No.
Type IVA battery adapter kit	1414863
Type VIA battery adapter kit	1414865
Type IC battery adapter kit	1414917

The secondary batteries contain positive electrodes of nickel hydroxide, negative electrodes of cadmium hydroxide and use potassium hydroxide as the electrolyte. The secondary batteries are rechargeable. Tables 15-5 and 15-6 give the battery characteristics.

The Type VIA secondary batteries are used in the self-destruct system. (See Par. 15.2.1.2 for battery capability.) The Type V secondary batteries are used in the command destruct system. The output voltage of the Type V secondary battery is nominal 28v dc. The battery is capable of a one-second surge into a 3-ohm load immediately after discharging 0.35 ampere-hour through a 200-ohm resistor. The battery terminal voltage during the surge will not be less than 18.0 volts.

15.3.1.2 Power Conversion. No optional equipment is supplied to augment the basic power conversion system.

15.3.1.3 Power Distribution. The following equipment is available to supplement the basic power distribution capability.

a. Ampere-hour Meter Kit (Spec 1414921). An optional ampere-hour counter and sensor device is available for use on programs desiring to monitor total current requirements. Installation may be accomplished without disturbing vehicle wiring; "readout" is accomplished via telemetry. This information may be used to assist in mission operational decisions and provides data for use in postflight analysis.

b. Safe and Arm Plug Kit (Spec 1414579). These kits provide safe and arm plugs as required, for use with program peculiar pyro circuitry to allow incorporation of the same pyrotechnic system testing and arming interface connectors as used on the Standard Agena. The safe and arm plugs consist of connectors with prescribed pairs of pins connected by means of jumper wires. The wired connectors are encapsulated in a can-type covering.

15.3.2 Command Destruct System

The basic self-destruct system must be replaced with the optional command destruct kit (Spec. 1414935) on vehicles launched at the Eastern Test Range to provide destruct capability up to orbit injection. Unlike the self-destruct system, the Agena command destruct system is self-contained in the Agena forward section and remains with the Agena throughout the mission. The command destruct system consists of two links which provide command destruct redundancy. Each link has a Type V secondary battery and a receiver-decoder which is coupled to two antennas by a multicoupler. The destruct system is initiated by a series of commands transmitted from the range safety transmitter and received by the destruct system antenna and receivers. The first sequence of commands is decoded and routed through relays to produce a 28v dc signal which disables only the Agena engine (engine inhibit). The second sequence of commands is decoded and routed to the command destruct unit which ruptures the Agena oxidizer and fuel tanks.

15.4 PROGRAM PECULIAR HARDWARE

A number of qualified electrical power system components have been developed to satisfy mission peculiar requirements. Their general characteristics, part numbers, and functions are discussed in pars. 15.4.1.1 through 15.4.2.

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Tables 15-2 through 15-4 list some equipment that is radiation resistant. This equipment performs without degradation when subjected to the following nuclear radiation environment.

Radiation Doses:

Neutrons with energy greater than 0.1 Mev — 1×10^{13} n/cm^2

Gamma photons of all energy — 1×10^{18} carbon rads.

Dose Rates:

Neutrons with energy greater than 0.1 Mev — 3×10^{7} n/cm^2/hr

Gamma photons of all energy — 1×10^{6} carbon rad/hr

Exposure time — 100 hours

Radiation Energy Spectrum:

10:1 ratio of fast neutrons (neutrons with energy equal to or greater than 0.1 Mev) to slow neutrons (neutrons with energy less than 0.1 Mev). The neutron to gamma ratio is approximately 10^5 n/cm^2/carbon rad.

15.4.1 Power System

Components have been developed by various Agena programs for use in the Agena electrical power system. These include primary and secondary batteries, power conversion and control equipment, and various electrical disconnects.

15.4.1.1 Power Source. Primary and secondary batteries other than those mentioned in pars. 15.2 and 15.3 have been developed to fulfill special mission requirements. The general characteristics are listed in Tables 15-4 and 15-5.

15.4.1.2 Power Conversion and Transfer Unit. Table 15-2 lists some of the dc-dc converters and inverters that are qualified. Also listed are the Type IA single-phase power amplifier and a Type VII voltage regulator. The relay transfer units are listed in Table 15-3.

The primary difference in the dc-dc converters is the output capability. The voltage regulation is the same as for the Type IX (±28.3v dc ±1 percent).

The Type VII voltage regulator is designed for use with a thermo-electric input power source which has an open circuit voltage within the range of 0 to 69v dc. The output voltage from the voltage regulator is 28.5v dc ±2 percent over the load range defined as follows:

Any load from 200 watts to power maximum

Power maximum (regulator output-watts) = 14.5 (Vs - 28.5) for $29 \leq Vs \leq 69$

When Vs = open current voltage of the input power source (volts).

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The transistorized static electronic power amplifier, Type IA, delivers a nominal 115v, 400-cycle, single-phase output. The single-phase output may be synchronized with an output of a three-phase supply by feeding one phase of the three-phase supply into the amplifier. The frequency tolerance is a function of synchronous input signal frequency tolerance.

A single-phase, 2000 cps, ±1 percent Type IV-D inverter is available. The output voltage accuracy is 115v ±5 percent (0-150 watt) when the load power factor is kept at 0.8 minimum lag to a 0.95 lead maximum.

Type I dc-dc and Type III 115v, 400-cps transfer relay units have been qualified and may be used where redundant power conversion systems are used. If a failure occurs in the primary power conversion unit, the transfer relay unit detects the failure and transfers the electrical system to the stand-by unit.

The dc-dc transfer relay will initiate transfer from the first regulated power source to the second when either the plus or minus output deviates more than 2.8v dc ±0.75 from the nominal ±28.3v dc level for a minimum delay of 50 milliseconds. The transfer time will not exceed 110 milliseconds. If a failure should occur in the second unit, the relay will transfer to the first unit. The contact rating for the dc relay is 28v, 10-ampere resistive, and 115v, 10-ampere inductive for the ac control relay.

The 400-cps transfer relay will supply +28v dc unregulated voltage to the standby unit when the output of the primary unit drops to 60±5, rms voltage. (Relay dropout time — less than 12 milliseconds). Once the output of the standby unit reaches 90±5 volts rms the transfer relay switches (relay pick-up time — less than 15 ms) the 115v, 400-cps bus from the first unit to the second unit.

15.4.1.3 <u>Power Distribution</u>. Program peculiar wire harnesses are installed in the Agena as required to provide interconnection for the various items of program peculiar equipment.

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In addition, a program peculiar electrical interface can be provided if required at the junction between the spacecraft and the spacecraft adapter. This interface is comprised of standard electrical connectors for those missions that do not separate the spacecraft from the Agena. For missions on which the spacecraft and Agena are separated, the interface will be comprised of an inflight disconnect.

Several in-flight disconnect designs have been developed and flown successfully. For some applications, a squib-actuated, rotary spin-off disconnect connector is desirable. At other times, extremely low withdrawal forces are required, or lanyard actuation may be necessary.

Shown below are some of the variations which have been used recently:

	Total Pins	Gauge	Spec. No.
Squib-actuated rotary disconnect	55	20	1347315
Low force	59 & 4	20 & 16	1364469
Low force	9	20	
Lanyard-actuated	12	20	1618741 (Pin Type) 1618742 (Socket Type)
Lanyard-actuated	41	20	
Lanyard-actuated	55	20	

A variety of electrical services may be provided at the interface connector through various combinations of the basic, optional, and program peculiar equipment discussed previously. These include (but are not limited to) the following:

- +28v dc unregulated pyrotechnic power and return
- +28v dc unregulated power and return
- ±28.3v dc regulated power and return
- 115v, 400-cps, one-phase power and return
- 115v, 400-cps, three-phase power and return
- Spacecraft monitor or control functions which interface with AGE via the Agena's J100 umbilical
- Spacecraft telemetry signals which interface with Agena telemetry system

In utilizing the available electrical service, the spacecraft contractor must provide certain information to LMSC and must exercise certain restraints, including the following:

- Specify if function is squib-operated or has any other peculiarities
- Specify load (in watts or volt-amps) and duty cycle (in percent operating time that load is on the Agena bus)
- Specify type of power required (ac or dc, voltage, tolerance, regulation, frequency, power factor, etc.)
- Ascertain that no spurious signals from the spacecraft or wiring exist that will adversely affect Agena electronic equipment operation

Launch pad electrical service for the spacecraft may be provided through an electrical spacecraft umbilical. The "cole" connector qualified for use on the Mariner Mars Program can be adapted to other programs. This connector is identified by LMSC-1460647 for the airborne half and LMSC-1460646 for the ground half. The connector will accommodate 69 size 16 AWG wires and two coaxial cables. The plug and receptacle are of the self-aligning type so that critical connector elements are protected from damage. Figure 15-2 shows the application of the connector as used on Mariner Mars.

A temperature transducer mounted on the connector by means of a leaf spring attachment provides for measurement of internal shroud temperatures up to launch without adding flight hardware: LMSC-1546244 identifies this feature.

15.4.2 Destruct System

Gemini Program requirements have resulted in a peculiar application of the optional command destruct kit which is used in conjunction with the self-destruct system. The command destruct system is modified to provide only permanent engine shutdown capability after separation. The destruct charge and associated cabling is not employed in this design.

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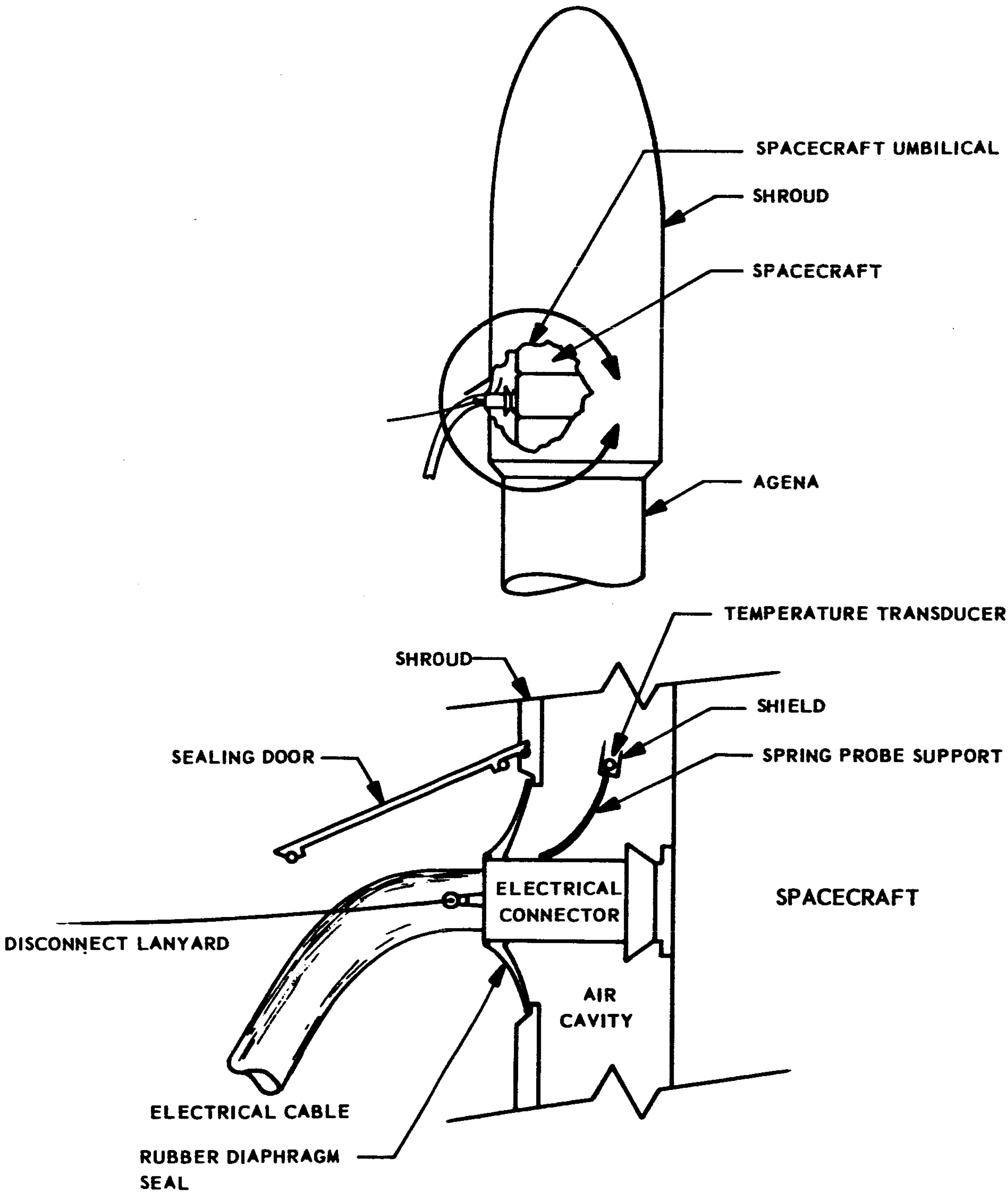

Figure 15-2 Spacecraft Umbilical Arrangement

Table 15-2

POWER SUPPLIES, INVERTERS (Qualified and in Use)

Nomenclature	Type	Output: Volts	Output: Watts (+/-)	Output: Volt-Amperes	Input	Weight lb	Dimensions L x W x H (inches)
dc-dc Power Supply 1461397	IX	±28.3v dc ±1%	60/20		22-29.25v dc	6.5	7.0 x 6.3 x 3.5
dc-dc Power Supply 1461411	X	±28.3v dc ±1%	350/15		22-29.25v dc	9.5	8.0 x 7.2 x 3.8
dc-dc Power Supply 1461466	XI	±28.3v dc ±1%	30/15		22-29.25v dc	5.0	6.5 x 5.7 x 3.0
*dc-dc Power Supply 1461635	XV	±28.3v dc ±1%	60/20		22-29.25v dc	6.5	7.0 x 6.3 x 3.3
1φ 400~Power Amplifier 1461173	IA	115v ±2%	0-10		22-29.25v dc plus 115v ±2% 400 cps	8	7.0 x 6.0 x 3.8
		115v ±1%	11-100		Sync Input		
*3φ 400 cps (±0.02%) Inverter 1461633	XI	115v ±1% (balance)	0-100		22-29.3v dc	17	10.9 x 9.7 x 4.1
		115v ±1.5% (balance)	101-200				
3φ 400 cps (±0.001%) Inverter 1463198	XII	115v ±1%	Up to 100% Unbalanced	0-200	22-29.3v dc	18	10.9 x 9.7 x 3.5
3φ 400 cps (±0.002%) Inverter 1464420	XIIA	115v ±1%	Up to 100% Unbalanced	0-200	22-29.3v dc	18	10.9 x 9.7 x 3.5
3φ 400 cps (±2%) Inverter 1464262	XIV	115v ±3%		15-70	22-29.3v dc	10	6.0 x 6.0 x 4.0
		115v ±10%		0-15			
*Voltage Regulator 1461414	VII	28.5v dc ±2%	585		0-69v dc	18	≈21.5 x 20.4 x 2
1φ 2000 cps Inverter 1461174	IVD	115v ±5%	250		22-29.3v dc	15	8.4 x 7.5 x 4.5

*Radiation hardened - see par. 4.3.5

Table 15-3

POWER CONTROL AND MONITORING EQUIPMENT

Nomenclature	Type	Characteristics	Weight (lb)	Dimensions L x W x H (Inches)	Remarks
Transfer Switch (motor reversible activated) 1062515	-	2 poles @ 60 amp 1 pole @ 5 amp	1.3	3.7 x 2.0 x 2.4	
Transfer switch (motor reversible activated) 1461797	X	2 poles @ 60 amp 1 pole @ 5 amp	1.8	3.5 x 3.7 x 2.9	
*Transfer Relay dc-dc 1461601	I	3 PDT @ 28v dc, 10 amp resistive	2.5	3.7 x 3.0 x 1.7	Senses ±28.3v dc - detects failure in primary system and transfer to backup system
Transfer Relay 400 cps 1461396	III	1 PDT @ 28v dc, 10 amp resistive 2 PDT @ 115v ac 10 amp inductive	2	3.5 x 3.0 x 1.8	Senses 115v 3∅ 400 cps - detects failure in primary system and transfer to backup system
Ampere Hour Meter System	-				
Counter 1461415-3		0-640 amp-hr capacity	3.3	3.8 x 6.2 x 3.1	3 channel TLM output @ 0-5v dc output signal to counter
Sensor 1461415-5		0-100 ampere dc	1.1	2.3 x 2.8 x 2.8	

*Radiation hardened - see Par. 4.3.9

Table 15-
PRIMARY BATTERY

Primary Battery Type	Remarks	Spec No.	Approx Amps	Calculated Int. Resist. Ohms	Oper. Temp Range (°F)	Wet Stand Life (Days) @ 30°F	Type of Cell
IC 1461791	Medium Rate	1414766	80	0.05	30-100	30	Ag_2O_2-Z
*ID 1461793	Medium Rate	1414730	80	0.05	30-100	30	
*IE 1461836	Medium Rate	1414446	80	0.05	30-100	30	
IVA 1464437	Highest Rate Battery Available	1067084	300A/ 0.25 SEC	0.02	40-100	7	
*VA 1461387	Remotely Activated, Uses Igniter Squibs	1412309	10	0.8	0-100 (with heater)	N/A	
*VI 1062762	Low-Rate, High-Efficiency	1067431	50	0.17	40-100	30	
*VI Mod 1336157	12 Volt Version of Type VI	1067431	50	0.085	40-100	30	
VI A 1461198	High-Rate Battery	1410550	80	0.03	30-100	18	
*XV 1463197	Radiation Resistant Version of IC	1415518	80	0.05	30-90	90	Ag_2O_2-Z

Table 15
SECONDARY BATTERY

Secondary Battery Type	Remarks	Spec. No.	Approx Amps	Calculated Int. Resist. Ohms	Oper. Temp. Range (°F)	Wet Stand Life at 30°F
V 1463136	Command Destruct System	1414343	7.5	2.0	30 to 125	2.5 years in discharge condition
VI A 1463150	Self-Destruct System	1415285	3	0.6	30 to 100	
*VII 1461649	Radiation Resistant - High Rate (Solar Array)	1461649	30	.05	0 to 125	
*IX A 1357358	High Rate	1415592	3	.2	0 to 125	2.5 years in discharge condition
*III 1062095	Formerly used in the destruct system (two in series)	1067050	3	0.3	30 to 125	

*Peculiar Batteries **99.5% Probability

4
CHARACTERISTICS

	No. of Cells	Nominal Volts	Ampere Hours	Watt-Hr Mean	Watt-Hr (**)	Max Wt (lb)	Watt-Hr Per lb	Maximum Overall Dim. L x W x H (Inches)	Volume
n	16	25	472	11,800	10,700	118	100	15.95 x 11.31 x 8.03	1450 in^3
	16	25	364	9100	8100	110	86	15.95 x 11.31 x 8.03	
	16	25	472	11,800	10,700	119	99	15.87 x 13.62 x 8.01	
	18	28	13	380	340	16	24	11.32 x 4.78 x 4.67	253 in^3
	16	25	15	400	-	13.5	30	9.82 x 4.90 x 6.89	
	17	26.5	70	1,980	1,450	27	73	12.53 x 6.25 x 5.70	
	16	12	130	1,980	1,450	27	73	12.53 x 6.25 x 5.70	
	17	26.5	46	1,320	966	27	49	12.53 x 6.75 x 5.70	446 in^3
n	16	25	400	10,000 (Est)	9,000	117	90	15.95 x 11.31 x 8.03	

-5
CHARACTERISTICS

	Charge Rate Amps	Type of Cells	No. of Cells	Nominal Volts	Amp Hours	Watt-Hr (mean)	Max Wt (lb)	Watt-Hr Per lb	Max Overall Dimension L x W X H (Inches)
a d	.040	N_i - cd	24	28	.35	10	3	-	7.08 x 5.03 x 2.30
	.045		6	7.5	.18	1.35	15 oz	-	4.00 x 2.73 x 1.50
	8		20	25	32	750	72	10	18.07 x 8.50 x 7.00
a d	1		20	25	6	140	15	-	11.4 x 5.8 x 4.8
	0.045	N_i - cd	3	3	0.225		5 oz		2.786 x 1.211 x 1.211

2

SECTION 16
GUIDANCE AND CONTROL SUBSYSTEM

16.1 GENERAL

The basic Agena guidance and control subsystem, illustrated in Figure 16-1, performs the vehicle guidance, control, and flight programming functions necessary to accomplish the vehicle mission. During the flight phases following separation of the Agena from the first-stage booster, the guidance system generates attitude error signals whenever the vehicle deviates from the prescribed attitude references. These signals are applied to the control system which utilizes pneumatic and hydraulic control forces to effect and maintain the proper attitude positioning of the vehicle.

16.2 OPERATIONAL MODES

The Agena is capable of two modes of active attitude control; one for fine attitude accuracy and the other for coarse attitude accuracy with minimum control gas consumption. Typical applications of the fine attitude control modes are in providing high pointing accuracy for spacecraft spin-up and separation, and for stabilization prior to engine operation. The coarse attitude mode can be used on missions which require extended coast periods. In addition, two possible modes of passive attitude control are available; one involves spin stabilization of the Agena about the pitch axis and the other uses gravity gradients for stabilization in a nose-down (or nose-up) orientation.

16.2.1 Active Control Mode

The Agena can be maintained in a constant attitude (within the limits of the deadband) with respect to earth over a period of hours or days with the basic

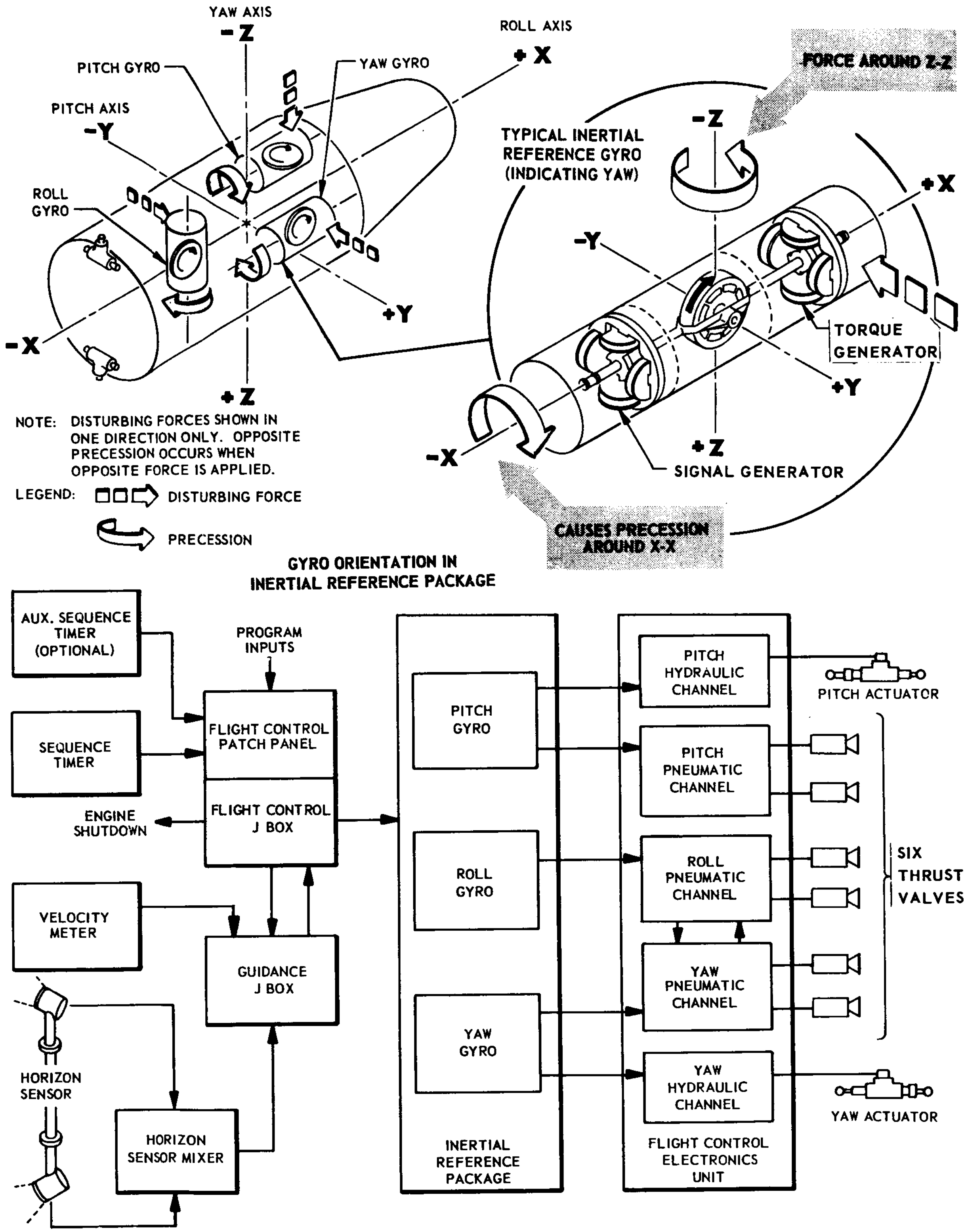

Figure 16-1 Guidance and Control Subsystem

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Agena guidance and control equipment discussed in par. 16.3. A command system must be added to implement the control system if, as described in the following paragraphs, a high accuracy control mode is required after sequence timer shutdown.

The high-accuracy control mode is initiated by giving commands for high-gain, high control gas pressure, and narrow deadband. High-gain operation increases the horizon sensor signal level to the gyros for fast positional error correction, and increases the pneumatic attitude control channel signal levels for fast response to the gyro control signals. High gas pressure (100 psi, 10-lb thrust per valve) torques the vehicle at a faster rate than low gas pressure (5 psi, 0.5-lb thrust per valve), and the narrow deadband command produces limit cycling signals within narrower limits of off-null deviations than the wide deadband command.

Some vehicle attitude deviations are permitted before corrective torques are applied. Two deadbands are available within the range of 0.2 to 8 degrees. One deadband permits only a narrow range of attitude deviation and thus maintains the vehicle in a fine or accurate attitude mode (±0.25 degrees typical). The other deadband permits a wide range of attitude deviation and thus maintains a coarse but low-gas-usage attitude mode (±5 degree typical).

The gas consumption (GC) rate is partially a function of the limit cycling. The limit cycle amplitude is constrained by the deadband. The limit cycle period, which is the time the vehicle takes to cycle once through the deadband limits, is dependent on the width of the deadband, pneumatic pressure, moment of inertia, and flight control gains. The number of thrust valve pulses (n) required to send the vehicle in the opposite direction at each end of the deadband varies with vehicle parameters such as moment of inertia, deadband and gain. Roughly, limit-cycling GC is inversely proportional to the width of the deadband and directly proportional to n squared. Total gas consumption is a function of vehicle maneuvers, disturbance torques, and total times in various control modes (fine or coarse attitude mode).

Though electrical power requirements may be reduced by de-energizing equipment (e.g., velocity meter, instrumentation) additional batteries may be required for extended missions. By placing the Agena in the deactivate mode (as discussed below), gas and electrical power consumption can be minimized.

16.2.2 Deactivate/Reactivate Mode

A means of conserving Agena electrical power and control gas is provided by the deactivate technique which can be used for extended-duration missions. In the deactivated mode, the Agena is spin-stabilized in the pitch plane to provide an equalized thermal gradient over the vehicle to protect low-temperature-sensitive equipment.

The deactivate sequence is controlled by a preset timer, which is started by real-time ground command. The pitch rotation during the deactivate mode requires no additional stabilizing devices provided that proper pitch and yaw moment-of-inertia relationships are established.

Shortly after initiation, the timer switches the control system to the ascent mode (high-pointing-accuracy), the orbit pitch rate is removed from the pitch-gyro torquer, and a 3 deg/sec pitch rate is applied. Power to the guidance and control equipment is then removed. The low-power command receiver stays on to allow the vehicle to be put in the reactivate sequence by real-time ground command at some later time.

Reactivation is accomplished by applying power to the guidance and control system which is programmed to remove the 3 deg/sec pitch rate and apply geocentric pitch rate. Within a short period, the horizon sensors will view

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the earth and reorient the vehicle properly. Flight experience with this technique has proven that no search mode is necessary for re-establishing earth reference.

This feature can be used on Agena missions lasting up to 30 days with 22 days in the deactivate mode. Several Agena vehicles using the spin technique have been successfully flown on 10-day missions.

16.2.3 Gravity — Gradient Stabilization

The Agena, when placed in a nose-down (or nose-up) position on orbit, is essentially gravity-gradient stabilized by virtue of its moment-of-inertia distribution. However, some form of damping is required to nullify oscillations which can build up to appreciable amplitudes as a result of the local gravity-field restoring torques. An active damping system employs control moment gyros (CMG) which damp out vehicle oscillations before they reach sizable amplitudes.

The Mod II system was specifically designed for this purpose and offers improved performance over the Mod I design, which was developed for use as a roll-yaw damper. In addition to possessing twice the angular momentum of the Mod I gyros, Mod II control moment gyros have an appreciably lower drift rate of 0.25 deg/hr, as compared to 1 deg/hr for the Mod I gyros, and therefore provide greater attitude accuracy. Two gyros are employed in a V-configuration to damp out oscillations directly about two axes, and to damp out oscillations in the third axis by second-order effects.

The use of this concept allows a long-life attitude control with a minimum power requirement. The program peculiar and optional hardware required to implement the various guidance and control modes is described in pars. 16.3 and 16.4 respectively.

16.3 BASIC GUIDANCE AND CONTROL EQUIPMENT

The basic Agena guidance and flight control system consists of guidance equipment, flight control equipment, and flight programming equipment. This equipment is described in the following paragraphs.

16.3.1 Guidance Equipment

The basic guidance system detects the attitude of the vehicle, initiates signals to control vehicle attitude and flight direction, and controls the periods of rocket engine operation. The system components include an inertial reference package (IRP), a horizon sensor, a velocity meter, and a guidance J-box. The guidance components are packaged in a guidance module (Fig. 16-2) located in the forward lower section of the vehicle, its underside forming a part of the exterior surface of the vehicle.

16.3.1.1 Inertial Reference Package. The inertial reference package (IRP) is the primary attitude-sensing component of the guidance system. Contained within its temperature-controlled interior are three single-degree-of-freedom, rate-integrating gyroscopes, each individually oriented so that it senses the angular displacement of the vehicle about one of its three major axes. The gyros used are two hermetic integrating gyro (HIG) units to sense pitch and yaw and one miniature integrating gyro (MIG) unit to sense roll.

The functions of the IRP are as follows:

a. To accept input signals from the horizon sensor and the program rate circuitry and to change the IRP reference attitude in accordance with these input signals.

b. To detect the difference between the attitude of the vehicle and the IRP reference attitude, and to generate an error signal with an amplitude that is proportional to the difference in attitude.

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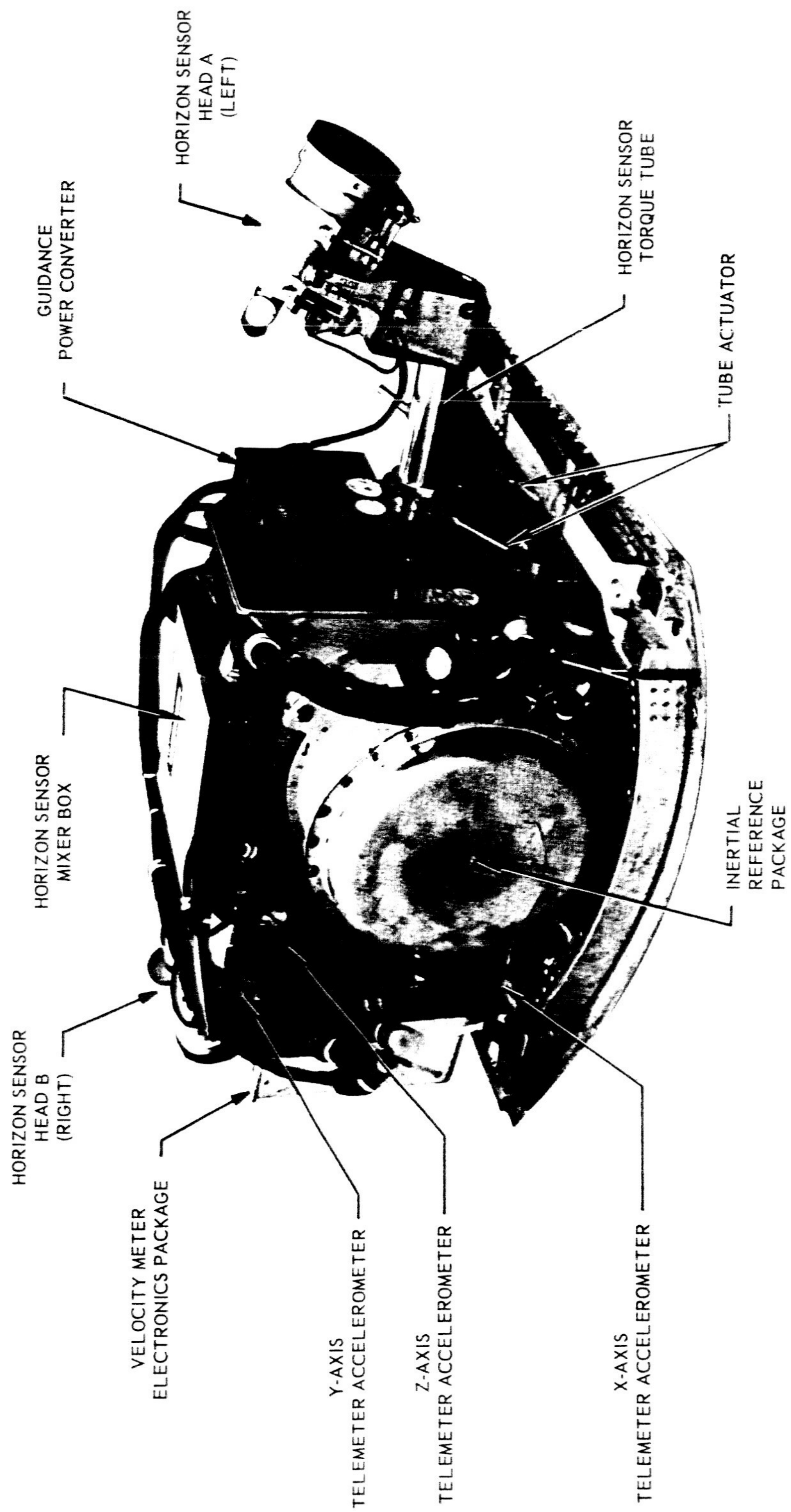

Fig. 16-2 Guidance Module

16.3.1.2 Horizon Sensors. The Barnes Mod IIC infrared horizon sensor (H/S), the second attitude sensing component of the guidance system, consists of two sensors and a signal mixer. Each sensor head scans the space below the vehicle, detects the discontinuity in the infrared (IR) radiation between earth and space, and generates a corresponding output signal. The signal mixer analyzes these two signals and develops a pitch attitude error signal and a roll attitude error signal whenever the vertical axis of the vehicle does not intersect the center of the earth.

The signal mixer does not develop a yaw attitude error signal because changes in the yaw attitude of the vehicle do not disorient the vertical axis of the vehicle from the earth's center. Therefore, these changes will not affect the IR discontinuity as seen by each sensor. A yaw reference is established from the booster attitude at burnout. Errors in yaw attitude during the coast period can be detected by the gyrocompassing circuits after a smoothing time of 10-15 minutes. This circuit applies the horizon sensor roll channel output signal and the roll gyro output signal to the input circuit of the yaw gyro. A corrective torque that is equal in magnitude but opposite in direction to the drift torque is developed and applied to the yaw gyro gimbal, thereby compensating for the drift torque.

Gyrocompassing functions to orient the vehicle to an orbit plane yaw reference are accomplished by detecting a component of the programmed pitch rate through the roll horizon sensor. The geocentric pitch rate (typically 4 deg/min for a low-altitude orbit) is programmed as a constant torque to the pitch gyro. This pitch component, if a yaw error is present, will be sensed differentially when compared with the roll components of the two horizon-sensor heads. The differential will be seen as a function of yaw error. The roll output is separated into a constant yaw-induced error and a normal roll oscillation by integrating over a time period. The roll-error component due to yaw is fed as a torquing signal to the yaw gyro, the output of which then corrects the vehicle's yaw orientation.

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A circuit in the mixer unit minimizes the effect of "cold" clouds which have tended to cause erroneous signals in earlier horizon sensors. If the horizon sensor sees a full sun in space or on the horizon, the signal generated will be the same as that of space. A spurious-signal detector circuit will inhibit the H/S head signal if the horizon sensor field-of-view grazes the sun in space. However, if the sensor field-of-view grazes the sun on the horizon, a horizon sensor error may be created due to the fact the sensor cannot differentiate between earth and sun. The theoretical maximum expected error is less than 1.7 degrees. Whether this error is sufficient to adversely affect the mission is dependent upon the trajectory accuracy required and also the time period in which the problem occurs. The critical periods are from Atlas separation through the first burn and from 600-second (nominal) time period prior to and through engine second burn. The above problem may be eliminated by restricting the launch window, or may be minimized by inhibiting the horizon sensor head by use of a sun detector device (par. 16.5.2).

The Agena is generally programmed to fire the engine in a slightly nose-up condition during first burn. This is done to ensure an optimum trajectory in the event of minimum boost apogee. When second burn is used, the bias angle of the horizon sensors is set to zero by a pyrotechnic device.

16.3.1.3 Velocity Meter. The velocity meter is the third major sensing element of the guidance system. Its purpose is to terminate the thrust of the rocket engine after the vehicle has increased its velocity by a predetermined increment. The velocity meter does not measure velocity; instead, it measures acceleration (change in velocity) and integrates this quantity to determine velocity.

The Bell Aerospace Corporation velocity meter consists of a pulse-output accelerometer and a pulse counter. The accelerometer senses acceleration along its sensitive axis and produces output pulses at a rate that is proportional to the sensed acceleration. The pulse counter counts these output pulses and energizes the thrust termination circuit when the total count reaches the pre-selected binary number that has been set into the counter.

16.3.1.4 Guidance Junction Box. The guidance J-box provides a distribution center for the signals and voltages entering and leaving the guidance module. It also functions as a central point for routing signals between the various components within the guidance module. In general, it is made up of relays and terminal boards mounted in a single chassis.

16.3.2 Flight Control System Equipment

The flight control system controls the vehicle attitude and direction of flight. Elements of the control system displace the vehicle about its three axes in response to signals from the guidance system. The flight control equipment consists of a flight control electronic unit, a pneumatic control system, a hydraulic control system, and a flight control junction box.

16.3.2.1 Flight Control Electronic Unit. The flight control electronic unit processes the attitude error signals from the IRP and distributes them to the control system. Processing consists of modifying and amplifying the signals and directing them through five electronic channels to actuate the pitch, roll, and yaw mechanisms of the pneumatic control system and the pitch and yaw mechanisms of the hydraulic control systems. The gas thrust valves are connected in pairs to three pneumatic channels. Each hydraulic actuator is controlled through a single hydraulic channel.

16.3.2.2 Pneumatic Control System. The pneumatic control system exerts control forces on the vehicle by release of cold gas through thrust valves (attitude control, etc.) to produce three-axis corrective torques. The system consists of six thrust valves in two clusters, a pneumatic regulator, and a control gas storage sphere. The location of the pneumatic system hardware and the thrust valves required for correcting various attitude errors are depicted in Fig. 16-3. The nozzles have a thrust rating of ten pounds each. Various gas mixtures of nitrogen and freon are used depending on the mission requirement.

The pneumatic system is activated at booster/Agena separation and provides pitch, yaw, and roll attitude control. At any Agena ignition, the pneumatic system continues to provide roll control, but the pitch and yaw controls are transferred to the hydraulic control system. At engine shutdown the pitch and yaw control is returned to the pneumatic system.

For orbital missions of short duration, the IRP, horizon sensor, and gas jets can be used to provide active attitude control as described above. For long duration missions, other flight-proven techniques of attitude control are available for the Agena (par. 16.2).

16.3.2.3 Hydraulic Control System. The hydraulic control system guides the vehicle during periods of engine operation. (See Fig. 16-4.) Directional control in pitch and yaw is accomplished by gimballing the rocket engine thrust chamber by means of hydraulic actuators controlled from the flight control electronic unit. Hydraulic power for the actuators is supplied from a hydraulic power package driven by engine fuel pressure.

16.3.2.4 Flight Control Junction Box. The flight control junction box with its integral flight control patch panel is the principal junction point for distribution of electrical signals within the guidance and control system. The flight control patch panel provides the means of varying the interconnection of the guidance module, flight control electronics unit, sequence timer, optional timer (if used), payload and program peculiar functions, telemetry, umbilical, and vehicle test points.

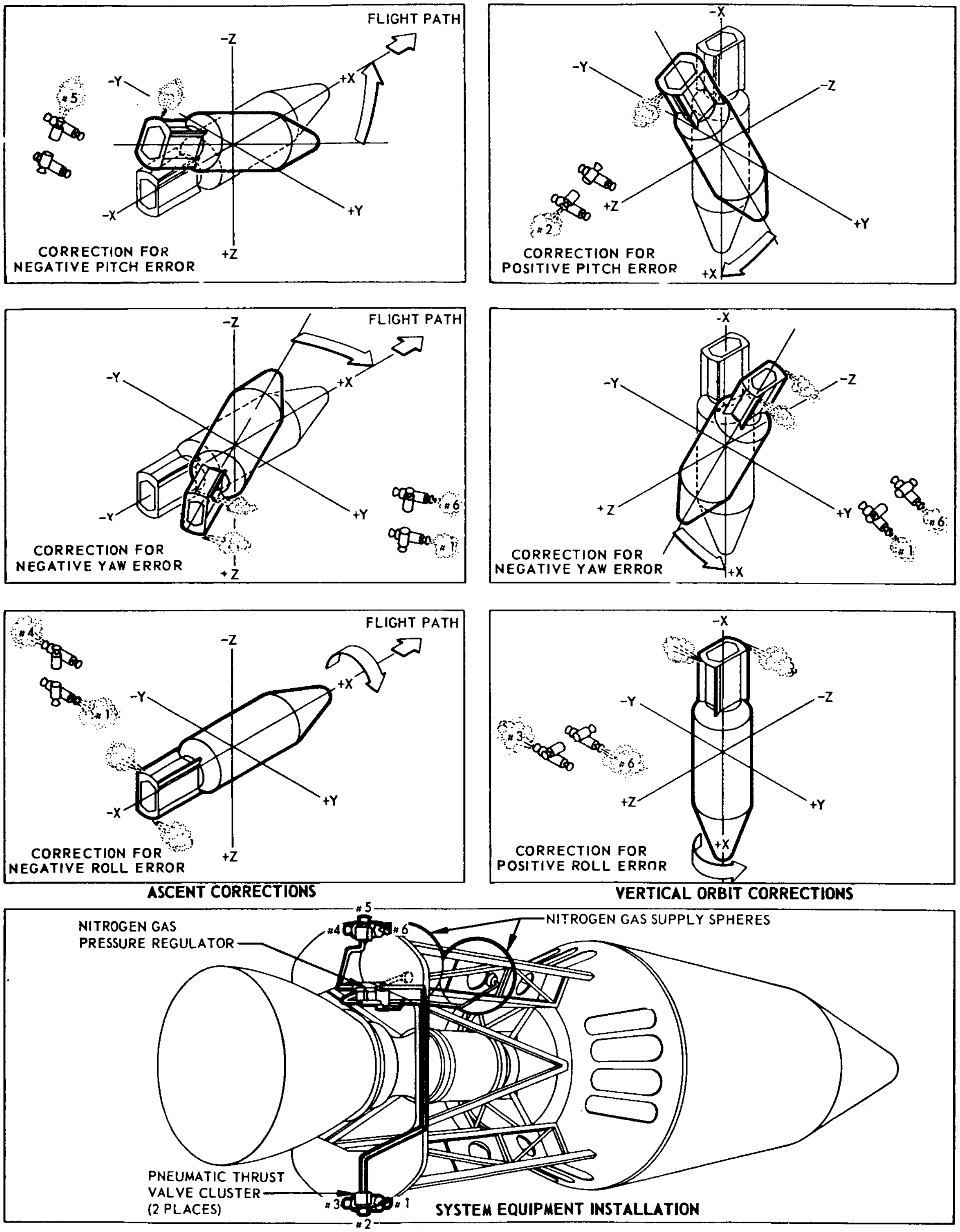

Figure 16-3 Pneumatic Flight Control System

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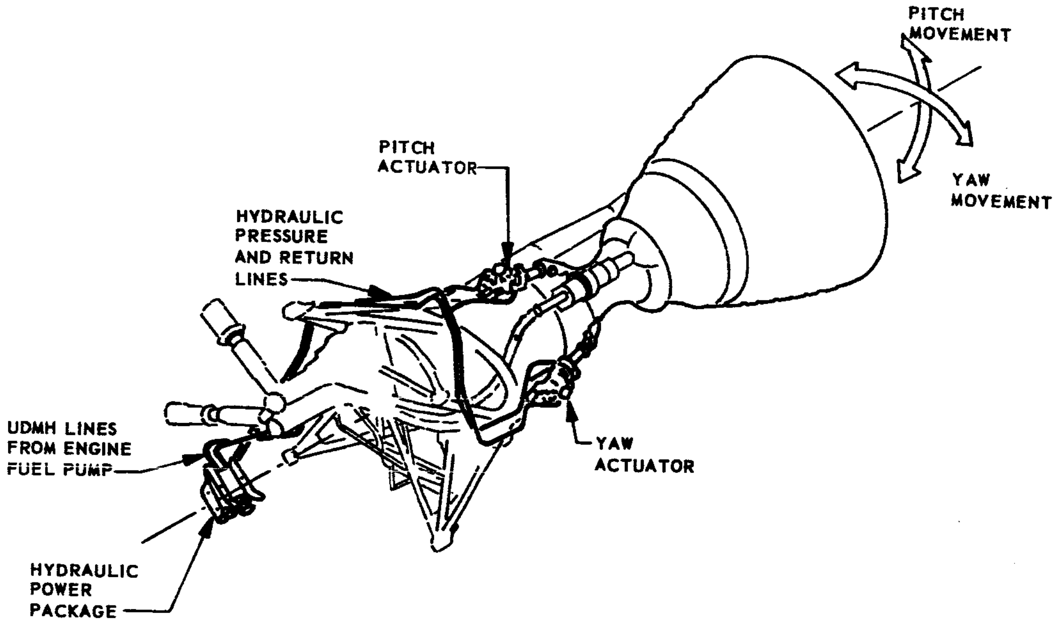
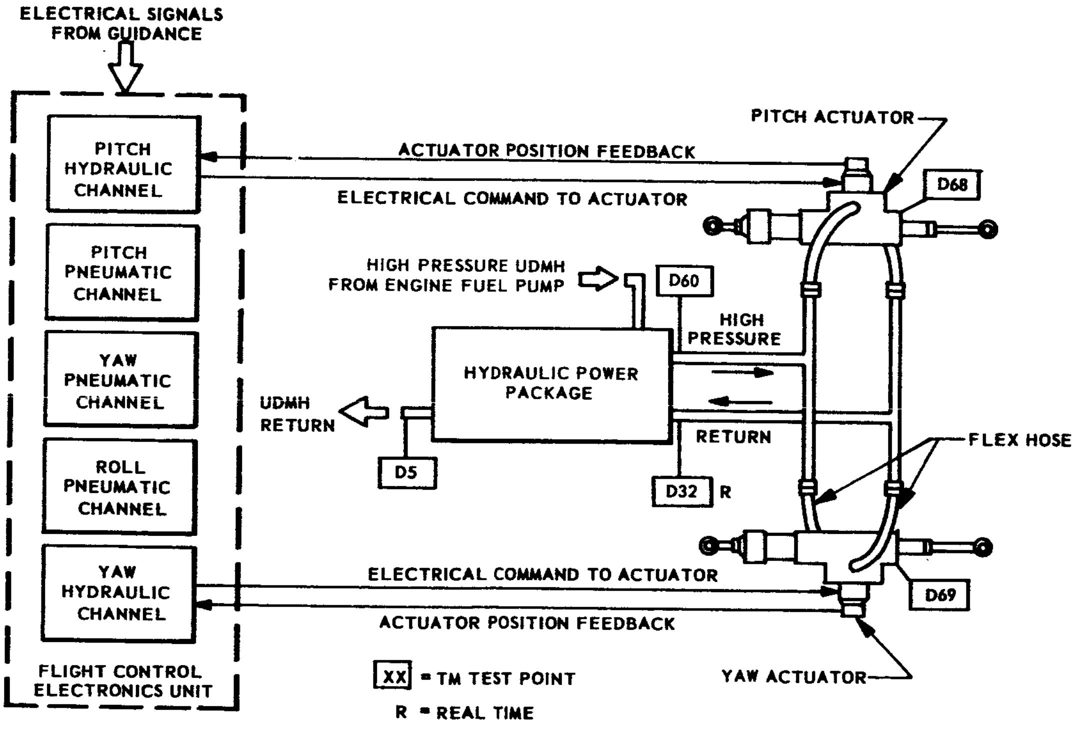


COMPONENT LOCATIONS.



SCHEMATIC DIAGRAM

Figure 16-4 Hydraulic Flight Control System

16.3.3 Flight Programming Equipment (Sequence Timer)

Flight event programming necessary to accomplish the Agena mission is provided by the primary sequence timer. Commands from the booster guidance system start the primary sequence timer (Atlas flights) or reset it (Thor flights). A secondary timer of similar function and design is also frequently used for backup, extended sequencing, or pre-programmed orbital maneuvers.

The ascent sequence timer (motor-driven mechanical timer) is used by all Agena vehicles for sequencing ascent events. The timer consists of 72 switches (10 amp resistive) arranged in 12 discrete switch banks consisting of six switches and are subdivided into groups ranging from two to four switches each. With this arrangement 24 separate events may be programmed. The majority of the switches are used by the Agena itself. Approximately one-third of the switches are brought to the flight control patch panel and utilized for variable mission functions.

The sequence timer can furnish the spacecraft with switch closures, +28v dc, or 28v return commands. It can also program Agena pyrotechnic power to initiate spacecraft pyrotechnics.

The timer accuracy is a function of power supply frequency, accuracy, and running time. This relationship is:

$$e_t = 0.15 \text{ sec} \pm \frac{e_f}{100} t = 0.15 \pm 0.00002\, t \text{ (sec)}$$

where

e_t = absolute magnitude of total difference between measured and set events (in sec)

e_f = percentage error of frequency from 400 cps = 0.002 percent

t = timer running time (in sec)

Assuming that the spacecraft event is required at the end of timer capability (6000 sec), the maximum time error is 0.27 sec. The interval between events is a multiple of one second.

If more complex programming is required, additional timers and programmers, as described in par. 16-4, may be added to the Agena in conjunction with the basic Agena timer.

16.4 OPTIONAL GUIDANCE EQUIPMENT KITS

Optional kits which are applicable to the guidance and control system are described in the following section. These kits have been developed and qualified under the Agena D (S-01B/SS-01B) program and are supplied by the Agena D program with the basic vehicle.

16.4.1 Flight Control Patch Panel Kit

This kit is required by all using programs to replace the basic vehicle test patch panel installed in the flight control J-box. Wiring of the patch panel is a using program responsibility. Installation of the programmed flight control patch panel provides program peculiar guidance and control signal distribution and power control routing points plus routing vehicle command functions. LMSC Specification 1414856 controls the design and manufacture of this kit.

16.4.2 Sequence Timer Kit

An unwired 6,000-sec sequence timer and necessary mounting hardware is contained in the kit. The timer is physically the same as the basic timer discussed in par. 16.3.3, except the switches are not wired. The timer is wired by the using programs in accordance with their requirements. LMSC Specification 1414861 applies to this kit.

Agena vehicles used on missions with extended sequencing (i.e., probes or extended orbital life missions) frequently require this kit for programming and sequencing of events beyond the capability of the ascent timer.

16.4.3 Sensor Bar Pin-Puller Kit

The installation of the pin-puller kit (LMSC Specification 1412937) provides a means of orienting the horizon sensors with the vehicle in the nose-down (or nose-up) attitude while in orbit, i.e., gravity gradient stabilization mode. The pyrotechnic actuated pin-puller permits the spring-loaded horizon sensor bars to rotate approximately 90 degrees.

16.4.4 Auxiliary Nitrogen Tank Kit

The auxiliary nitrogen tank kit is used to install a 2200 cubic inch capacity high pressure flask in the aft equipment support structure. The auxiliary tank provides a supplemental source of control gas to prolong the service life of the attitude control system. A temperature sensor is supplied for measuring gas temperature. When used in conjunction with an existing pressure transducer, it furnishes the capability for telemetering accurate information on the gas consumption. LMSC Specification 1414832 applies to this kit.

16.4.5 Horizon Sensor Pre-Amp Signal Conditioner Kit

Installation of this signal conditioner kit provides the using program with a capability for monitoring and telemetering of the horizon sensor pre-amp outputs. This kit may be installed without disturbing the basic vehicle wiring. LMSC Specification 1414875 applies to this kit.

16.4.6 BTL Adapter Kit

The BTL adapter kit is required to accommodate installation of the Bell Telephone Laboratory (BTL) command guidance system in the Agena D. The kit consists essentially of the BTL skin, VHF stub antenna, wire harnesses, BTL umbilical door, BTL control package, ventral fairings, fairing covers, and necessary mounting hardware. The BTL-600 guidance canister which contains the integrated command receiver and transponder

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circuits, RF commanded steering relays, and Ledex switch along with the peripheral waveguide assemblies, directional couplers, and ventral and dorsal antennas are Government Furnished Equipment (GFE) and not part of this kit.

The Agena installed BTL radio command system may be used to provide the TAT/Agena with increased guidance accuracy during ascent. The BTL system is described in Section 5. LMSC Specification 1414913 applies to this kit. Vehicles utilizing the command destruct kit cannot use the BTL adapter kit.

16.5 PECULIAR GUIDANCE EQUIPMENT

Program peculiar guidance equipment that is of potential value to a spacecraft contractor is listed in the following paragraphs.

16.5.1 Control Moment Gyro (Model II)

Application of the control moment gyro (CMG) is discussed in par. 16.2.3. LMSC Specification 1414177B establishes the design requirements.

The Model II control moment gyro is a single-degree-of-freedom, rate-integrating gyro. It consists of a gimbal and spin-motor assembly, electrically driven torque generator, and a signal generator.

16.5.2 Sun Detector Assembly

A horizon sensor head inhibit device may be required to minimize the effect of the sun on the horizon sensor system. (See par. 16.3.1.2.) A fully qualified and flight proven horizon sensor head inhibit device is available. The sun detector assembly (LMSC-1346133) consists of a cadmium-selenium detector, transistors, resistors, a relay, and a potentiometer; all enclosed in a cylindrical body. The field-of-view of the detector extends 2 degrees on either side of the horizon sensor field-of-view. Thus it "sees" the sun just before the horizon sensor does and disables the head just before the sun enters the sensor's field-of-view. The device removes the horizon sensor roll output from the IRP roll gyro and grounds the gyro torque input line during the inhibiting period.

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SECTION 17
COMMUNICATION AND CONTROL SUBSYSTEM

17.1 GENERAL

Transmission of data signals from the Agena to ground stations is a major function of the communications and control subsystem (C&C). The communications and control system also receives, decodes, and processes command signals from ground stations and, in some instances from the spacecraft. Because of the specialized nature of the C&C system, the equipment offered as basic equipment with the Agena is minimal. A range of optional equipment is required to support the particular mission. The choice of these optionals depends upon the ground station configuration supporting the mission. Paragraphs 17.3 (optional hardware) and 17.4 (program peculiar hardware) describe the qualified equipment available as "building blocks" for the C&C system.

The Agena normally carries a Type V FM/FM telemetry system as illustrated by the block diagram in Fig. 17-1. Approximately 85 points in the Agena vehicle systems are monitored during ascent to determine vehicle performance which usually requires 7 continuous channels and two commutated channels. Eleven channels may be added to the basic system to telemeter payload functions, or to obtain additional environmental data.

The functions of tracking, acquisition, and command of the Agena vehicle are performed by a beacon transponder, an acquisition transmitter if required, and an orbital programmer (if required). The beacon function may be performed by either the optional S- or C-band systems. A VHF acquisition transmitter is available for early acquisition of the Agena. The orbital programmer controls the various satellite orbital functions.

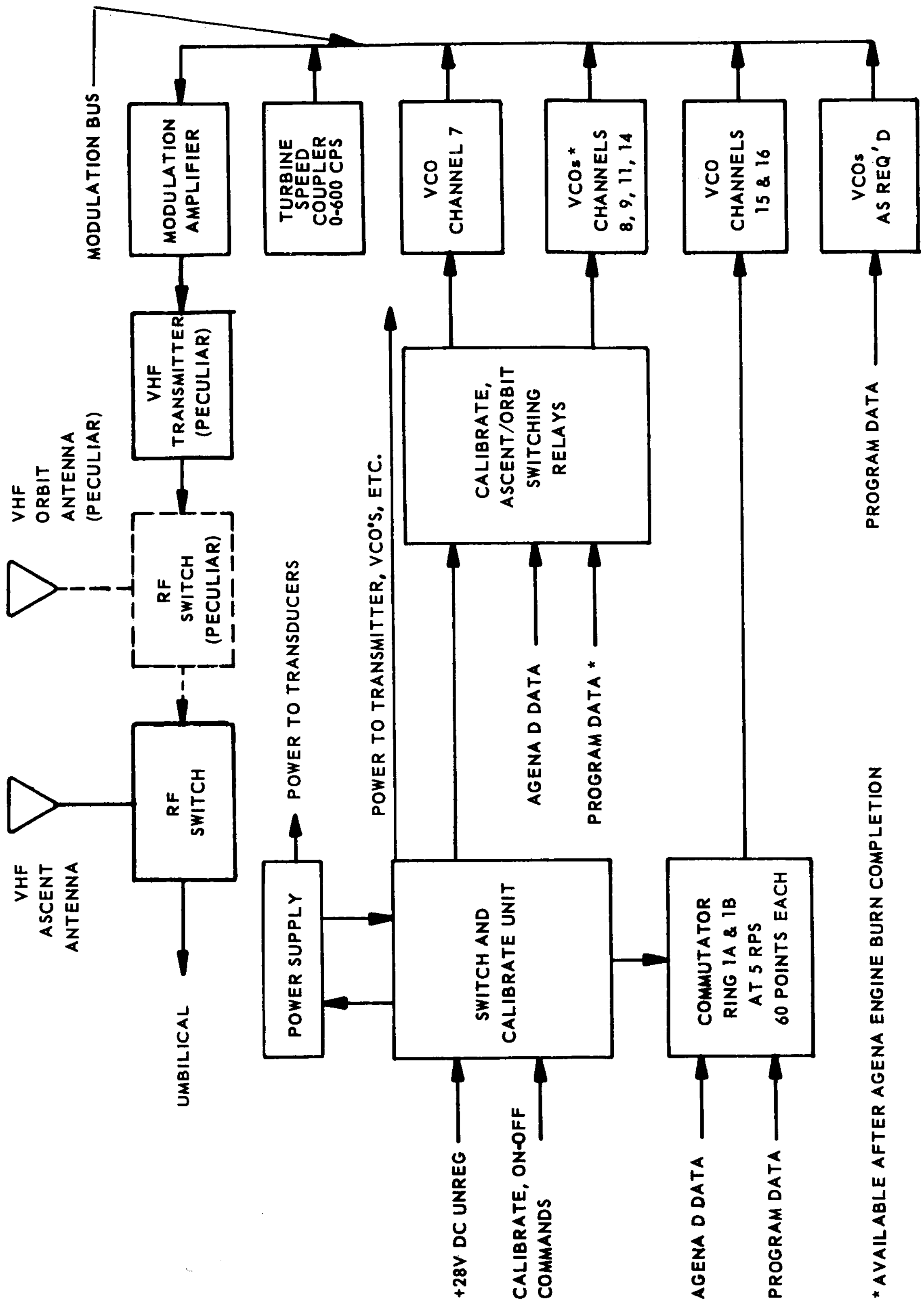

Figure 17-1 Telemetry, Type V Block Diagram

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17.2 BASIC COMMUNICATION AND CONTROL EQUIPMENT

The basic C&C equipment on the Agena is essentially the Type V FM/FM telemetry system. Optional kits for installing the various command or tracking systems are available to the using program. The telemetry system consists of various transducers, the Type V FM/FM telemetry module, a VHF RF switch, a VHF antenna and RF umbilical connector pins. The RF switch (Type II) used in the S- or C-band tracking system is also installed on the basic Agena.

17.2.1 FM Telemetry Module (Type V)

The Type V telemetry module is comprised essentially of one communication panel on which the transducer power supply, wiring harnesses, and an FM telemetry unit are mounted. The telemetry unit consists of a motor-driven commutator (two wipers, 60 points each), a switch and calibration unit, an oscillator tray, an oscillator input unit, and seven voltage-controlled oscillators (VCOs). The IRIG subcarrier VCO channels are 7, 8, 9, 11, 14, 15, and 16 with the commutated data carried on channels 15 and 16. Two VCOs other than channel 2 may be installed on the basic unit by the using program.

The basic Agena telemetry has 43 data input points with a sampling rate of five per second (60 points at 5 rps) available for program use. For data requiring higher frequency response other IRIG channels, except 2, 7, 8, 9, 11, and 14, may be added to the basic telemeter module. Channels 8, 9, 11, and 14 are available for program use after completion of Agena engine-burn functions.

Optional transmitters, VHF Type IV (2 watt) or VHF Type V (10 watt) may be utilized in conjunction with the Type V telemetry system. (See par. 17.3.1 for details.)

Prior to launch, the RF output from the transmitter is either transmitted to the ground by hardline or by the VHF antenna. The Type IV RF switch provides the means for transferring the transmitter RF output to the umbilical or to the antenna.

17.2.2 Radio Frequency Switch (Type IV)

The Type IV RF switch is a "fail-safe," single-pole, double-throw, non-latching solenoid switch. When RF output through the hardline is required, 28v dc is applied to the two solenoids via the electrical umbilical (J100). The holding power per coil is approximately 6 watts.

The following is the operating characteristic of the Type IV switch:

Cross talk:	30 db minimum at 3000 Mc.
Insertion loss:	0.2 db maximum at 3000 Mc.
VSWR:	1.2 maximum at 3000 Mc.

The Type IV switch is used by missions that require both ascent and orbit VHF antenna. In this application the solenoids are energized by the sequence timer switch and de-energized by the timer after the vehicle is injected into orbit.

17.2.3 VHF Ascent Antenna (Type XIII)

The E-slot-type VHF antenna with a flush-mounted window is installed in the skin surface of the forward section just aft of the guidance module on the +Z axis (earth side). The antenna is designed for use with a fifty-ohm transmission line, and is linearly polarized. The output of the acquisition transmitter is coupled with the telemetry VHF RF output and is radiated out through the same antenna.

17.2.4 Radio Frequency Umbilical Connector

The major electrical umbilical receptical (J100) has a special four-coax and 110-pin insert assembly. Two of the four coax connectors are used by the

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basic Agena, one for VHF telemetry system and the other is used by either the C-band or S-band tracking system. The remaining two pins are available for the spacecraft or program peculiar RF system.

17.2.5 Radio Frequency Switch (Type II)

The RF Type II switch is essentially an RF Type IV switch enclosed in a pressurized aluminum container. The input VSWR is less than 1.3:1 at 3000±20 Mc in either position with a matched 50-ohm input.

A coax cable from one of the coax pins on the J100 umbilical is connected to the RF switch. The other two coax cables are installed by either the optional C- or S-band beacon adapter kit.

17.2.6 Aft Instrumentation Box

An instrumentation box is installed on the aft rack of the Agena. The box distributes power to the aft pressure and temperature transducers and signal conditions the measurements.

17.3 OPTIONAL HARDWARE

Optional equipment is necessary to complete the basic Agena communications and control (C&C) system. Provisions are made for installing particular telemetry, tracking, command, and programmer equipment. The above equipment and associated hardware such as coax cables, RF switches, mounting brackets, etc., are installed by optional kits.

Various combinations of RF equipment can be installed with a minimum of engineering design effort. Figure 17-2 is a schematic diagram showing the different combinations.

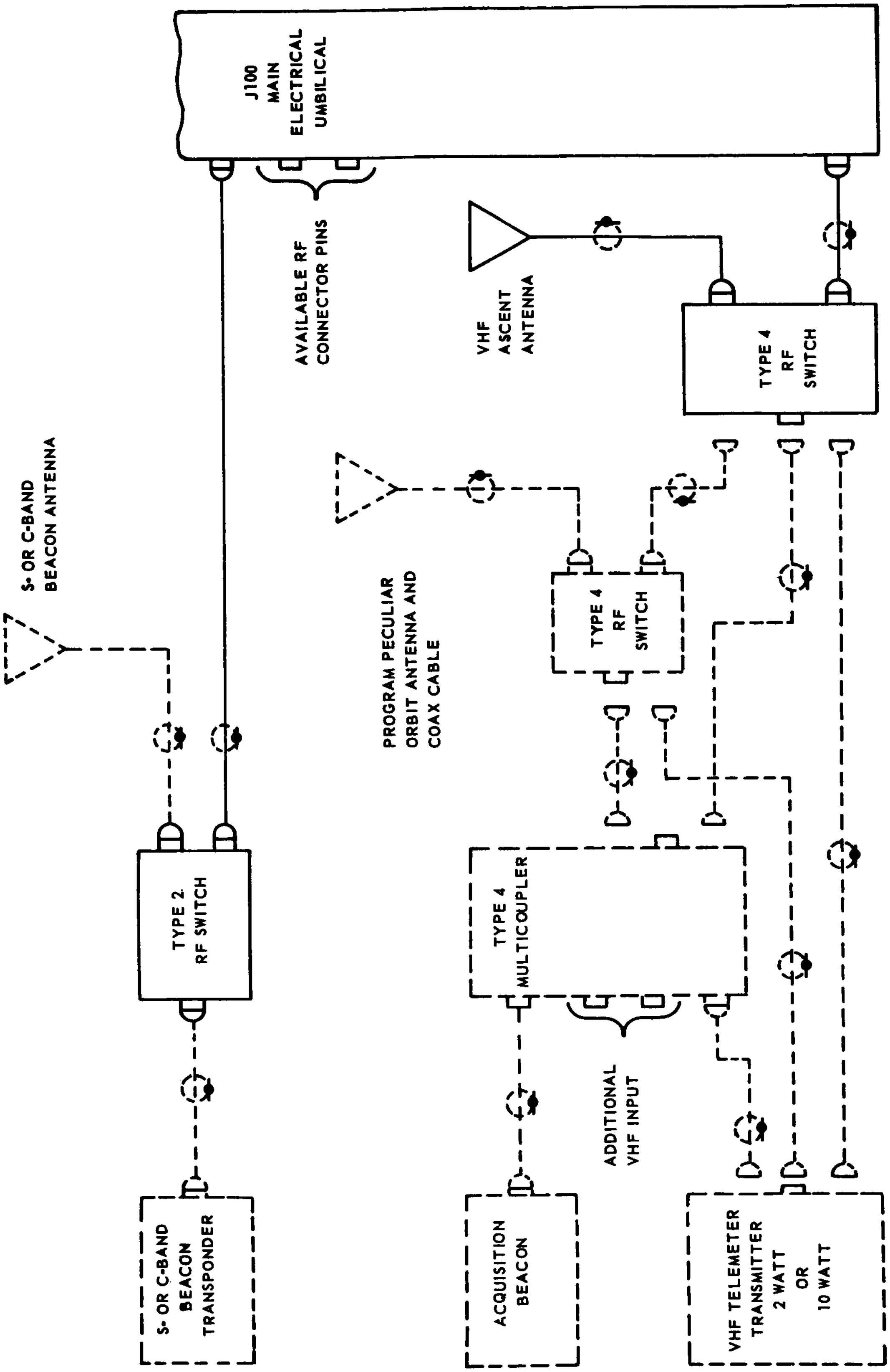

Figure 17-2 Schematic Diagram - Basic and Optional RF Switches and Associated Coax Cables

17.3.1 VHF Transmitters (Telemetry)

Either the VHF transmitters Type IV (2 watt) or Type V (10 watt) may be mounted on the Type V telemetry module described in par. 17.2.1. The 2-watt transmitter is normally used for earth orbit missions and the 10-watt transmitter is used on space probe missions, i.e., Mariner- and Ranger-type missions.

The Type IV transmitter (LMSC-1461400) has the following characteristics:

Power input:	+22 to 29.25 volts at 750 ma maximum
Power output:	1.8 watts minimum into a 50-ohm resistive load with 22 to 23.5 volts applied
	2 watts minimum into a 50-ohm resistive load with 23.5 to 29.25 volts applied
Frequency:	Tune-fixed at any frequency between 225 Mc and 265 Mc
Frequency stability:	± 0.01 percent (long term)
	0.005 percent (short term, approximately 30 minutes)
Input impedance:	10K ohms minimum
Modulation input:	5 to 125,000 cps flat within ±1.5 db
Deviation (±):	Selectable, 120 ± 10kc, 60 ± 5kc, 30 ± 2.5 kc
Deviation sensitivity:	50kc/volt, 25 kc/volt, and 12.5 kc/volt

The Type V transmitter (LMSC-1461341) has characteristics similar to the Type IV except for input-output power:

Power input:	Plus 22.0 to 29.25v dc at 3.3 ampere maximum
Power output:	9.0-watt minimum into 50-ohm load with 22 to 23.5 volts applied, 10.0 watt minimum into 50-ohm load with 23.5 to 29.25 volts applied

Provision for installation of the transmitters on the telemetry module is provided by optional TLM transmitter adapter kits. The two transmitter adapter kits (2 watt and 10 watt) are designed per LMSC specifications 1414581 and 1414582, respectively.

Programs which require VHF RF output to be radiated from a program peculiar orbit antenna use the TLM orbital antenna switch kit. (See Fig. 17-2.) The kit consists of a Type IV RF switch, coax cable, and mounting hardware. The design and performance of the kit is per LMSC specification 1414591.

17.3.2 Voltage-Controlled Oscillators

Voltage-controlled oscillator (LMSC-1461717), IRIG channels 2 through 18, and A through E are available.

A non-IRIG channel F VCO (98kc frequency response of 2940 cps), LMSC-1464013, is also available. The VCO input parameters are as follows:

The input impedance:	one meg-ohm ±20 percent for channel 2 through 18, and 500,000 ohms (minimum) for channels A through E
Source impedance:	one ohm to 10,000 ohms
Input voltage:	0 to 5v dc
Modulation sensitivity:	plus or minus 7.5 percent of center frequency for IRIG, Band IRIG channels A through E

As many as eleven VCOs may be added to the basic Type V Telemetry module, two on the basic VCO tray and nine on the auxiliary VCO tray which is a part of the auxiliary FM TLM adapter kit (Type V). VCOs 2, 3, 4, 5, 6, 10, 12, 13, 17, 18, E, and F may be installed but channel 2 may not be used during ascent. The using program may modulate the optional oscillator channels with either a continuous or commutated measurement. The design and performance of the kit is per LMSC specification 1414583.

17.3.3 PAM Telemetry System Type VIII

A fully qualified type VIII PAM telemetry system (LMSC-1396636) module is an alternate package which is interchangeable with the FM/FM Type V package.

Pulse amplitude modulation (PAM) is one type of time-division multiplexing whereby information to be transmitted is sequentially sampled. The sampling produces a series of pulses with the amplitude of each pulse being an indication of the data amplitude. The sampling procedure utilized by the multiplexers employs electronic gating techniques to provide the necessary time sharing.

The PAM/FM system consists of a 16-channel main multiplexer (Type VI), a 128-channel submultiplexer (Type IV), two dc-dc power supplies and either a Type IV or a Type V VHF transmitter. LMSC-1414535 provides the detail specifications for the Type VIII PAM Telemetry System. (See Table 17-3 for total number of data input channels.)

17.3.4 Acquisition Beacon (Type IC)

A Type IC acquisition transmitter may be installed to provide for vehicle acquisition. The continuous wave (CW) output of the transmitter is multiplexed with the VHF telemetry RF and radiated through the basic ascent VHF antenna (Type XIII). The frequency range of the transmitter is between 216 and 265 megacycles. The power output is a nominal 10 milliwatts of continuously radiated VHF signal.

Two optional kits are usually required to install the acquisition beacon system, the acquisition transmitter adapter kit and the VHF multicoupler adapter kit. In cases where the C- or S-band modules are not used the blank communication panel kit may be required.

The acquisition transmitter adapter kit installs the Type IC acquisition transmitter (not part of the kit), coax cable, and wire harness on the beacon module. The design and construction of the kit is per LMSC specification 1414859.

17.3.5 VHF Multicoupler (Type IV)

A Type IV VHF multicoupler (LMSC-1062243) which is capable of accepting four simultaneous RF inputs and feeding such inputs to a common single output is available. The operating parameters are listed below:

Frequency:	can be tuned to various VHF frequencies
Minimum RF Isolation:	18 db for a 2.8 to 3.8 Mc separation between adjacent channels 20 db for a 3.8 Mc or greater separation between adjacent channels
Impedance:	input and output compatible with 50-ohm system
Input VSWR:	no greater than 1.5:1
Insertion Loss:	1.5 db or less
Power:	max. total = 60 watts max. per channel = 30 watts

The VHF multicoupler adapter kit provides the necessary hardware to complete the acquisition system. The kit consists of a 50-ohm coax termination, coax cable, and attaching hardware. The kit installs the four-channel, Type IV multicoupler (not part of the kit) on the beacon module. The coax cable links the multicoupler output to either the basic Type IV RF switch or to the Type II pressurized RF switch installed by an optional kit. (See par. 17.3.)

Two of the four multicoupler input channels are used by the basic VHF telemetry system and the acquisition beacon system. The remaining inputs are available for other VHF TLM links. The design and construction of the kit is in accordance with LMSC specification 1414590.

17.3.6 S-Band Beacon Transponder

The S-band beacon transponder (Model WDL-RT-18) operates in conjunction with the VERLORT radar at the ground sites to provide a means of automatically tracking the vehicle, and to provide for ground-to-vehicle communication when operated with the beacon decoder. The transponder characteristics are listed in Table 17-1.

The transponder consists of a receiver, a transmitter, and a diplexer which permits one antenna to function for both receiving and transmitting. Upon reception of a three-pulse code from the VERLORT radar, the transponder will determine if the code is correct, and if so, will respond with a short pulse of energy which provides the VERLORT with tracking information. The transmitted response also serves to identify a particular vehicle.

The transponder circuitry is so aligned that it will accept only the proper code pulse, all others being rejected. The transponder detects the coded pulse by sampling the first and third pulses of the three-pulse group, and if the spacing is correct, a trigger is generated which enables the transmitter to respond.

The second pulse of the three-pulse group is tone-modulated (pulse position modulated). This modulation causes the second pulse to vary its position relative to the first and third pulses. The second pulse has no effect on transponder operation and is used strictly for command coding.

The S-band beacon adapter kit, Type III, installs the S-band transponder. The kit consists of two coax cables, S-band beacon antenna, C-band beacon antenna cover and a beacon module structure panel on which the S-band transponder (not part of the kit) is installed. The kit is designed and constructed in accordance with LMSC specification 1414871.

17.3.7 S-Band Beacon Decoder (Type IX)

Fifteen real time commands are available when the Type IX decoder (LMSC-1461745 or Philco 95-142431) is used with the S-band beacon transponder. The beacon decoder is enabled whenever the coded pulse received by the transponder is correct. The coded pulse is routed from the transponder to the decoder. A gate which is in time coincidence with the third pulse is also supplied from the transponder. These two signals are utilized by the decoder to demodulate the second pulse of the three-pulse group.

The second pulse has been previously tone-modulated (PPM) by the VERLORT, this modulation causing the second pulse to vary its position relative to the first and third pulses. The modulating tone is a combination of two audio frequencies and represents one distinct command. A total of six frequencies, using all possible combinations of any two, result in a total of 15 commands.

17.3.8 C-Band Beacon Transponder (Type V Radar)

The C-band transponder (LMSC-1461702) is similar in operation to the S-band beacon transponder but is used for tracking only. The unit characteristics are listed in Table 17-2.

The C-band beacon adapter kit installs the C-band transponder. The kit consists of two coax cables, a Type II C-band beacon antenna, S-band beacon antenna cover, and a beacon module structure panel on which the transponder (not part of the kit) is installed. The kit is designed and constructed per LMSC specification 1414872.

17.3.9 Orbital Programmer, Type VIII and Type VII

A punch-tape type programmer (Type VIII) for command sequencing orbital functions is available. The programmer is a device capable of storing and actuating 52 off-on events (±2 seconds accuracy) per orbit (sub-cycle). It includes two separate units, one electronic package (LMSC-1461895), and a tape magazine and drive package (LMSC-1461895). The programmer drives the four tape decks. The memory is stored in the form of tape perforations. Each deck has the capacity to store 128 sub-cycles. The sub-cycle period is adjustable from 86.5 to 103.2 minutes (period range of a low earth orbit). The programmer utilizes real-time commands adjusting the period. It may be reset and may also be rapidly advanced or rewound.

A Type VII orbital programmer (LMSC-1461701) is also available. The operation is similar to that of the Type VIII except the electronics and the tape decks

(2 instead of 4) are in one unit. This unit will program 26 events with an accuracy of ±10 seconds. The sub-cycle range is from 87.3 minutes to 108.4 minutes.

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The support structures, mounting brackets, and interconnecting cables for the Type VIII orbital programmer are contained in the SS-01B-furnished orbital programmer mounting kits. The Type VII orbital programmer also uses the kit but does not require the interconnecting cables. The design and construction of the kit is per LMSC specification 1414914.

17.4 PECULIAR HARDWARE

A number of qualified communication and control systems components have been developed to satisfy mission peculiar requirements of several programs. The hardware is discussed in paragraphs 17.4.1 through 17.4.4.

17.4.1 Telemetry System

A Type I PCM telemetry system has been qualified for the Gemini Agena Program and is compatible with existing Mercury-Gemini ground stations. In the near future, it will also be compatible with the mini-track stations that are being modified for the Biosatellite Program.

The PCM-I System (LMSC-1346486) consists of two 128-channel multiplexers, a PAM-PCM encoder, a telemeter control unit, two Type IV transmitters, a Type IV RF switch, and a multicoupler. Two Type V transmitters can be substituted for the Type IV transmitters if the additional power output is required.

The PCM-I accommodates three broad categories of data: analog, direct digital, and pulse-rate analog. Analog data is sampled and encoded into a binary format for transmission. Direct digital exists in binary format and does not require encoding for transmission. Pulse analog is used for measurement of the gas generator turbine speed. The telemetry module counts and totals the number of pulses from the turbine speed transducer. This accumulated total pulse count is transmitted as a binary word.

Table 17-1

S-BAND TRANSPONDER SPECIFICATIONS

RECEIVER	
Frequency	2800 to 3000 Mc/sec tunable
Type	Superheterodyne
Sensitivity	Minus 65 dbm minimum over frequency range
Selectivity	2-cavity preselector — minimum of 20 db down at plus and minus 25 Mc from center frequency
Bandwidth	8 Mc/sec ±2 Mc/sec
Stability	±4 Mc/sec over temperature range of -30°F to 165°F
Spurious Response	Image 40 db down
TRANSMITTER	
Frequency	2800 to 3000 Mc/sec tunable
Power	Min. 1000 watts peak - Max. 2000 watts peak
Stability	±4 Mc/sec over temperature range of -30°F to 165°F
Pulse Width	0.8 ±0.1 microseconds
Pulse Rise Time	0.1 microsecond (10% to 90% points)
Repetition Rate	0-1600 pps
Delay Time	1 ±0.5 microseconds
Delay Variation	0.25 microseconds over 15-db range
Recovery Time	100 microseconds
POWER	
Input Voltage	22 to 30v dc
Input Power	Maximum of 45 watts

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Table 17-2

C-BAND TRANSPONDER SPECIFICATIONS

	RECEIVER
Frequency	5400 to 5900 Mc/sec tunable
Type	Superheterodyne
Sensitivity	Minus 65 dbm minimum over frequency range
Selectivity	2-cavity preselector — minimum of 25 db down at plus and minus 25 Mc from center frequency
Bandwidth	5 Mc/sec +2 Mc/sec
Stability	±2 Mc/sec over temperature range of -30°F to 165°F
Spurious Response	Image 40 db down
	TRANSMITTER
Frequency	5400 to 5900 Mc/sec tunable
Power	Minimum 500 watts peak — maximum 750 watts peak
Stability	±4 mc/sec over temperature range of -30°F to 165°F
Pulse Width	0.5 ±0.2 microseconds
Pulse Rise Time	0.1 microsecond (10% to 90% points)
Repetition Rate	0-1600 pps
Delay Time	2.5 to 3.0 microseconds
Delay Variation	0.10 microseconds over 15-db range
	POWER
Input Voltage	22 to 30v dc
Input Power	Maximum of 30 watts

A block diagram of the PCM system, as used in the Gemini/Agena Program, is shown in Fig. 17-3. See Table 17-3 for the number of input channels and sampling rates.

17.4.2 Tape Recorders

Magnetic tape recorders are in common use in Agena programs. Some of the tape recorders presently used in the Agena telemetry system are listed in Table 17-4.

17.4.3 Tracking Beacons

Acquisition transmitters and tracking transponders are available for use as qualified equipment. This equipment is discussed in Table 17-5.

17.4.4 Vehicle Command System

Agena vehicles used as orbital spacecraft have command systems ranging from simple to complex. Commands are received in the Agena through either the command receiver, such as the VHF Type I, the UHF Type IX, or through the S-band transponder. The demodulated commands are then provided to a decoder which checks for pulse length, spacing, and frequency, and transmits the command to the proper equipment in the case of real-time commands. For stored commands, where it is desired to execute the command out of sight of the ground station, a programmer is used.

The LMSC Type XVI programmer will be used on the Gemini-Agena program, and receives digital commands directly from the command receiver. After verifying the proper format, real-time commands are sent to the associated controller for execution. Stored commands are sent to the magnetic-core memory unit to await the designated time interval before they are transmitted to the controller for execution. The programmer contains a crystal-controlled electronic clock which is accurate to two parts in one million in a 12-hour period. The clock has a resolution of one second. The core

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memory unit is capable of nondestructive readout over telemetry for command verification. A number of subroutines (such as the engine-start sequence) which are initiated by a single command are built into the Type IV command controller, which must be used in conjunction with this programmer. The capability for both real- and stored-time commands is 93 with a memory capacity of 64 commands. A modification could be incorporated which would permit a total of 125 commands. The available command units are shown in Table 17-6.

17.4.5 Flight Instrumentation

A variety of flight evaluation instrumentation may be provided for any given mission, being limited only by performance capability, cost, and available telemetry channels. The instrumentation may be physically mounted on the spacecraft and serviced by the Agena telemetry, or it may be installed on the shroud and spacecraft adapter to provide information regarding the spacecraft environment.

A listing of several categories of instrumentation that have been provided on past missions is provided below. Typical TLM channel assignments are listed for each measurement to indicate Agena telemetry utilization.

Sound Microphones - Located within the shroud cavity to provide spectrum and intensity information regarding the acoustic environment within the shroud cavity during launch, transonic, and max q. (Continuous channel 18*.)

Accelerometers - Located in the spacecraft adapter to provide information regarding the low-frequency vibration environment at the spacecraft/Agena interface from launch through spacecraft separation. (Continuous channels 10, 12, and 13.)

Vibration - Located on the spacecraft adapter to provide high-frequency spacecraft vibration information. (Continuous channels 17, and 18*.)

*These channels are time-shared with other measurements.

Spacecraft Separation Monitors - Three separation monitors located at the spacecraft/adapter interface to measure linear separation velocity and "tip-off" rates. (Continuous channels 8*, 9*, and 11*.)

Shroud Internal Pressure - One transducer located within the shroud cavity at station 240 for measurement of internal pressure. (Commutated on channel 16.)

Spacecraft Temperatures - Located on critical spacecraft components or adjacent to them within the shroud cavity to provide on-pad and ascent spacecraft temperature data. (Commutated on channel 16.)

Spacecraft Pressures - Four spacecraft canisters within the shroud cavity to provide on-pad and ascent pressure data. (Commutated on channel 16.)

*These channels are time-shared with other measurements.

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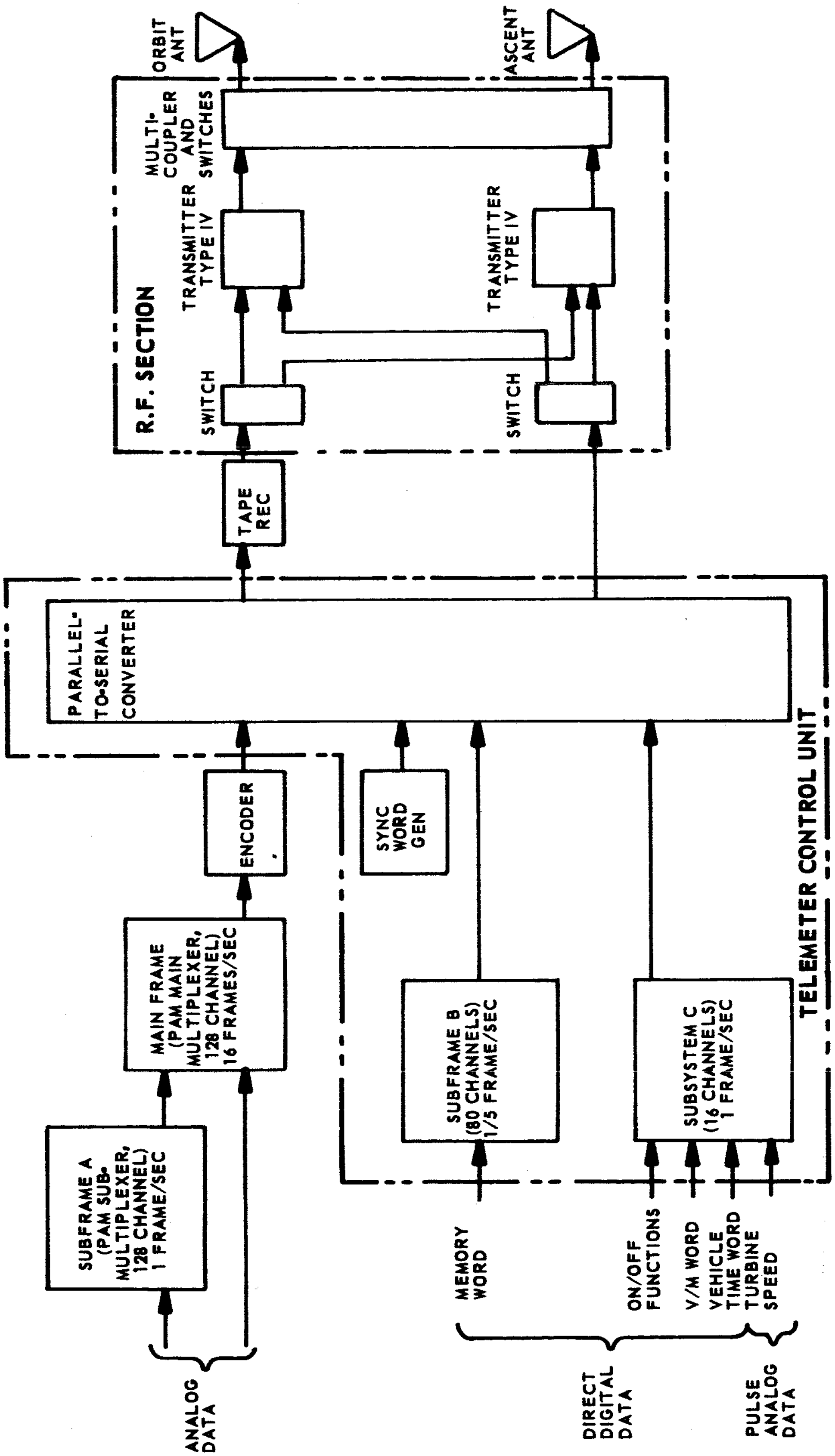

Figure 17-3 PCM Telemetry System Data Flow Diagram

Table 17-3

SUMMARY OF AVAILABLE AGENA TELEMETRY

Type, LMSC P/N	Characteristics	Power Ratings Input-Output (W)	Wt (lb)	Status
Telemetry				
FM/FM 1398396	7 continuous channels 2 commutated @ 60 points/sec Max. capability - 18 channels	29/2* or 80/10**	19	Qualified and flown
PCM-1 1346486	124 channels @ 1 frame/sec + 108 channels @ 16 frames/sec + 80 (24-bit digit) words @ 1/5 frame sec + 16 (8-bit digit) words @ 1 frame/sec	39/2* or 120/10**	29	Qualified to be flown in 1965
PAM-8 1396636	124 channels @ 19.5 frames/sec or 124 channels @ 4.0 frames/sec + 13 channels @ 2500 frames/sec or 13 channels @ 512 frames/sec	35/2* 90/10**	-	Qualified
Transmitters				
Cubic Corp 1461400 Type IV, 1461341 Type V	VHF (216-265 Mc) IRIG Intelligence input: 5 cps, 125 kc	19/2 or 100/10	2/4	Qualified and flown
LMSC Type 8 (UHF 1348362 & 1364917	2.2-2.4 kMc; wide band Base band response: 10 cycle to 5.5 Mc @ ±2 db 10 cycle to 7.5 Mc @ -3db	100/10	21	Qualified and flown

*With Cubic Type IV transmitter

**With Cubic Type V transmitter

Table 17-4
TAPE RECORDERS*

Type No.	LMSC Part No.	Frequency Response	Record Time (min)	Reproduce Time (min)	Power (W)	Wt (lb)	Remarks
IX	1357274	16384 bits/sec	20	5	15 (ave) 25 (peak)	11	Used with PCM/FM system Qualified - to be flown in 1965
VIII	1461799	128 cps	135	10.5	12 (Record) 25 (Reproduce)	9.75	Presently used with PAM/FM system Qualified - to be flown in 1965
XII-5	1461870	60 cps	180	15	6 (Record) 8 (Reproduce)	7.20	Used with FM/FM system Commutated data Qualified & Flown
XIII	1463199	200-50 kc	12	6	6 (Record) 8 (Reproduce)	-	Analog Type Qualified & Flown

*Partial list of tape recorders that are qualified and presently being used.

Table 17-5

SUMMARY OF AVAILABLE TRACKING UNITS

Type	Characteristics	Power Ratings Input-Output (W)	Wt (lb)	Status
Transponders				
Philco S-Band Type XI LMSC-1461745 Philco WDL 95-142431	(2.8-3 kMc) - also used with Type II decoder (tracking and command)	35 1000 (min.) Peak 2000 (max) Pulse	8	Qualified and flown
Motorola C-Band LMSC-Type VII LMSC-1464040	5690 Mc Receiver 5765 Mc Transmitter (Used for tracking only)	30 500 (min.) Peak 750 (max) Pulse	12	Flight-proven item undergoing minor modification - used in Gemini Agena mission
C-Band (Type V Radar) LMSC-1461702	(5.4-5.9 kMc) used for tracking only	30 500 (min.) Peak 750 (max) Pulse	-	Qualified and flown
Doppler Tracking & Command System LMSC-Type XI LMSC-1461833	(400 Mc band) - used as phase-lock system; capable of 32 commands (real time); command bit rate, 200 bits/sec	(14/0.032)	25	Qualified to be flown in 1965
Acquisition Transmitters				
VHF-Type I Philco 99-103005	(216-265 Mc) Stability limits ±0.001%	0.6/0.010 (CW)	1	Qualified and flown
UHF-Type I LMSC-1348389	(399-401 Mc Stability limits ±0.0035%	5/0.040	2	Qualified and flown

Table 17-6

SUMMARY OF AVAILABLE COM

Hardware	Characteristics and Remarks
Programmer	
Type VII (LMSC-1461701)	26 S. T. Cmmds* available with accuracy of ±10 sec
Type VIII (LMSC-1461895)	52 S. T. Cmmds available with accuracy of ±2 sec
Sequence Timer (LMSC-1341515)	24 events with total of 72 switches. Two of the above events used for presetting the timer. One second resolution with ±0. 5 seconds accuracy
Type XVI (LMSC-1348306)	93 R. T. cmmds* available max. command rate 15/sec 93 S. T. cmmds available - memory capacity at any one time is 64 commands - max. execution command rate 64/sec. Maximum loading command rate 5/sec This programmer is used with Type IX UHF receiving LMSC-1464003 which is qualified and is to be flown in 1965
Doppler Tracking & Command System Type XI (LMSC-1461833)	(400 Mc band) - used as phase-lock system; capable of 32 R. T. commands; command bit rate, 200 bits/sec
Command Controller	
Type IV (LMSC 1372700)	Used in conjunction with Programmer Type XVI to distribute command executions, provide sub-routines and reset timer.
Decoders	
Type XI (Philco 95-142431) LMSC-1461745	15 R. T. cmmds - this decoder is used with S-band beacon transponder (Philco WDL-95-140666)
Type V (LMSC-1461489)	R. T. cmmds; 1 relay closure plus 3 digital commands. The decoder is used with the VHF Type I Receivers
Type XII (LMSC-1461715)	4, 7, or 11 R. T. cmmds channels are available (relay closures). The basic unit has 4 channel output and by adding additional "Slave" decoders, the channels are increased. The decoder is used with the UHF Type I Receiver.
Type XIII (LMSC-1352015)	3 R T. digital cmmds. The decoder is used with the S-band beacon transponder (Philco WDL-95-140666)
Receivers	
Type IX (LMSC-1464003)	Used with Type XVI programmer UHF frequency
Type I (LMSC-1461488)	Used with decoders 5, 12

*S. T. Cmmds = Stored-Time Commands   R. T. Cmmds = Real-Time Commands

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MAND UNITS

Status	Input Power (W)	Wt (lb)
Qualified and Flown	10	13
Qualified and Flown	10	27
Qualified and Flown	17.5 Timer Motor 12 Timer Brake	8
Qualified - to be flown in 1965	8 watts exclusive of clock oscillator heater 9 watts oscillator heater maximum	43
Qualified - to be flown in 1965	14	25
Qualified - to be flown in 1965	Power derived from Programmer Type XVI	29
Qualified and Flown	6.5 maximum 30.0v dc input	7
Qualified and Flown	0.1 during standby 0.13 during interrogation	2
Qualified and Flown	0.003 during standby 1.0 during interrogation	4
Qualified and Flown	2.25	4
Qualified - to be flown in 1965	10 maximum	15
Qualified and Flown	0.85 (interrogate power)	13

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SECTION 18
LAUNCH BASE AGE AND FACILITIES

18.1 GENERAL

NASA Atlas/Agena satellite and probe launches from Cape Kennedy utilize Complexes 12 and 13 and associated facilities. NASA launches by Thor/Agena or TAT/Agena from Vandenberg Air Force Base utilize Complex 75-1-1 and its associated facilities.* Equipment and services available at these launch complexes are discussed in this section.

18.1.1 Pad Orientations

Launch pad azimuth orientations, with respect to the downrange (+Z) coordinate of the Agena is as follows:

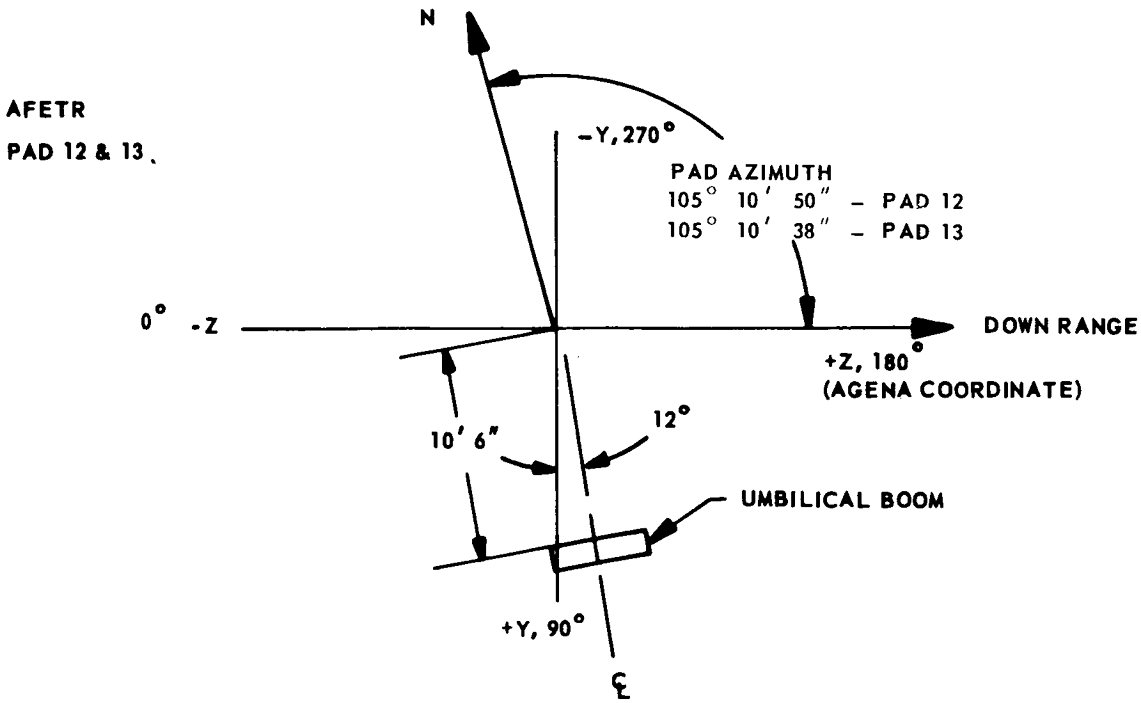
AFETR Complex 12	-	+Z coordinate 105° 10' 50" SE
AFETR Complex 13	-	+Z coordinate 105° 10' 38" SE
AFWTR Complex 75-1-1	-	+Z coordinate 259° 30' WSW

Additional details are shown in Fig. 18-1.

18.1.2 Gantry Work Platforms and Hook Heights

Both Cape Kennedy and VAFB gantries provide protective enclosures for the spacecraft and the forward end of the Agena. At various levels within these enclosures there are platforms available for working access to the launch vehicle. Platform locations and hook heights are shown in Figs. 18-2, 18-3, 18-4. Umbilical connections to the spacecraft and Agena can be made through the ends of the gantry enclosures.

*Atlas/Agena launch capability existing at Point Arguello, near Vandenberg AFB, has not been applied to NASA use. TAT/Agena capability has not been developed at Cape Kennedy.



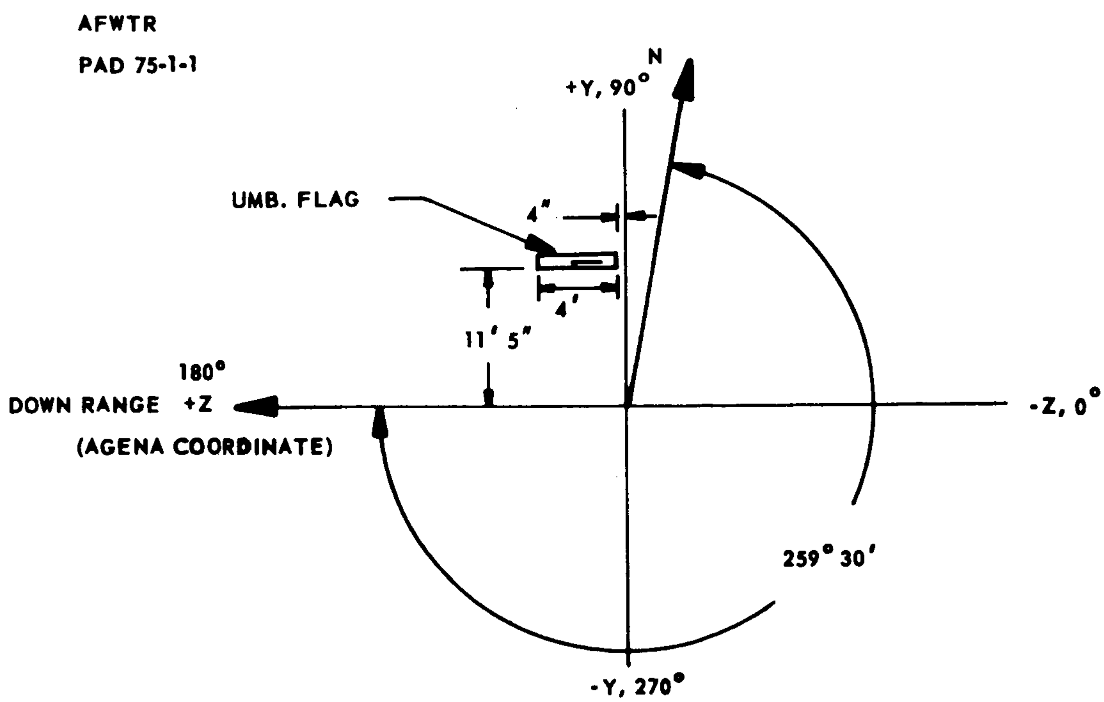


Figure 18-1 Pad Orientation Plan Views, ETR & WTR

E-3236-5

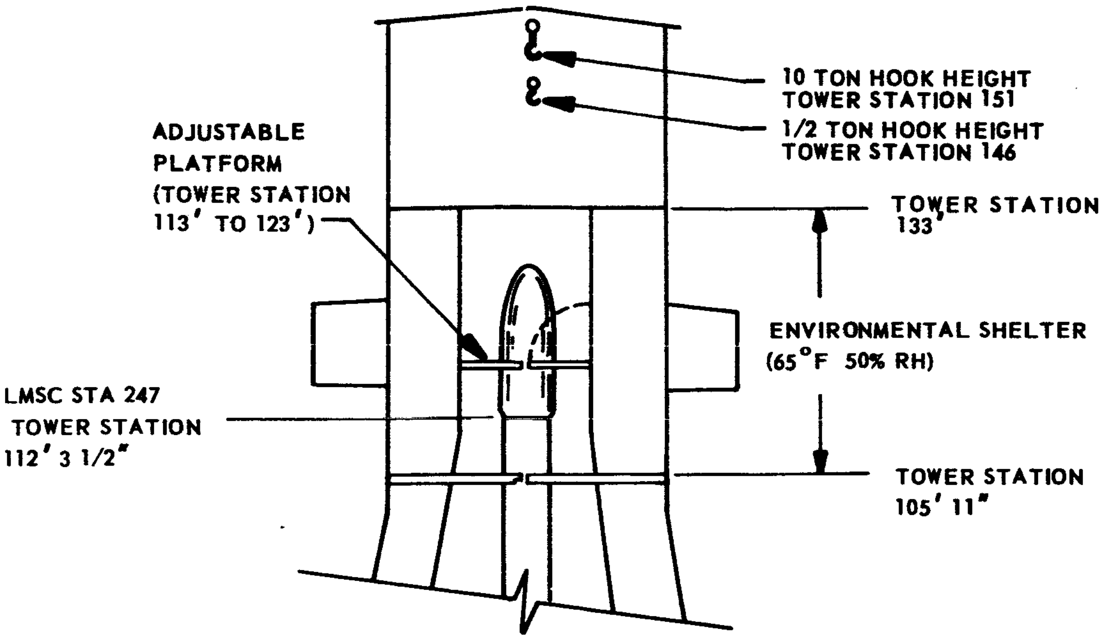


Figure 18-2 Pad 12 Gantry

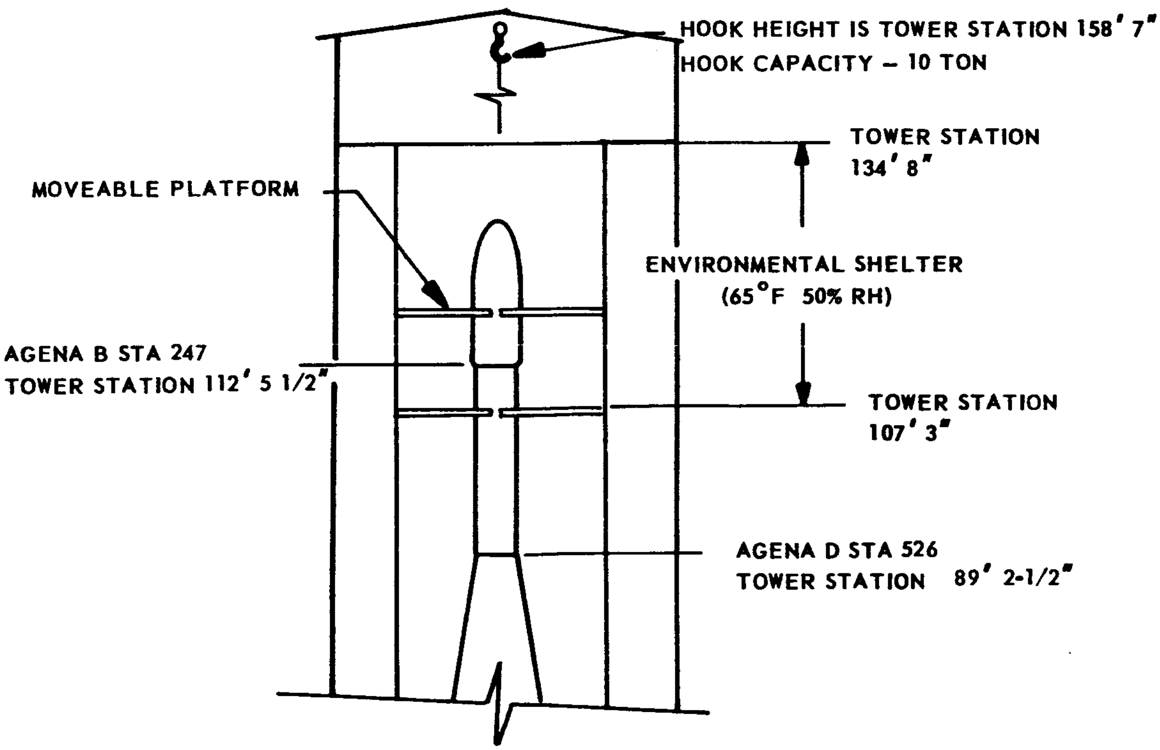


Figure 18-3 Pad 13 Gantry

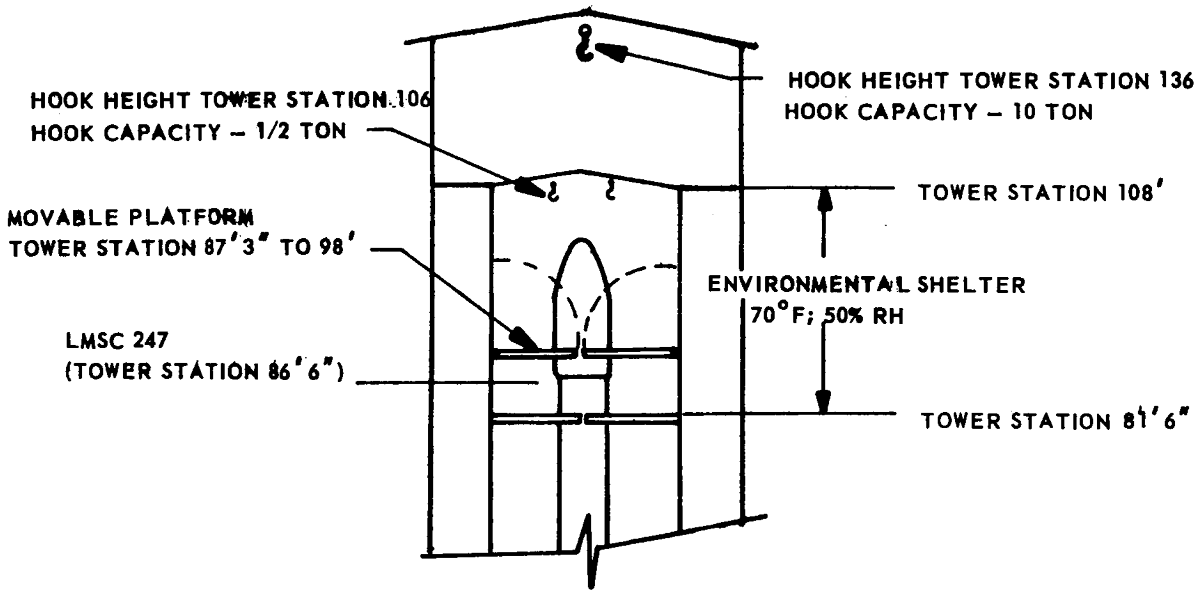


Figure 18-4 Pad 75-1-1 Gantry

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18.2 LANDLINE LAUNCH COMPLEX CONNECTIONS FOR SPACECRAFT

The Agena umbilical wiring is capable of handling an external power source of 28v dc for supplying the spacecraft while on the pad. Twenty-six No. 16 gage wires are allotted for spacecraft circuitry via the Agena's J-100 umbilical wiring. As the launch complexes are modified for successive programs, new landlines may be added for spacecraft use. For this reason, the number of lines available at the time of any proposed future use should be verified by the user. Present launch complex spacecraft support wiring is shown in Figs. 18-5, 18-6.

18.3 LAUNCH CONTROL SYSTEMS FOR AGENA AND SPACECRAFT

The Launch Control Systems (LCS) consist of the recorders, monitors, controllers and transmission lines for signals going to and from the Agena D vehicles.

The location of the LCS equipment is in rack-mounted panels in the Launch Operations Buildings (LOB) and the Launch Pad Buildings (LPB). The majority of the monitor and control equipment exists in the LOB while the LPB contains power supplies, signal conditioners, interconnect boxes and signal distribution consoles. Figure 18-7 through 18-12 show the space expected to be available for spacecraft contractor use for specialized LCS equipment.

Included in Agena LCS is the RF capability required to check the Agena's command transmission. A TV system for remote monitoring of the pad areas for observation of the umbilical disconnect is also part of LCS equipment.

18.4 GROUND SERVICE EQUIPMENT (GSE)

All fixtures, plumbing, connectors, and ducting necessary to service the vehicle is considered GSE. The function of GSE is to pressurize the vehicle during countdown using inert gases, load the vehicle with propellants, transport the propellants, and provide air conditioning when required. Air conditioning is provided to the Agena and the spacecraft during checkout to maintain the specified internal environments in the Agena forward rack and within the shroud cavity.

18.4.1 Vehicle/AGE Umbilical Connections

At each launch complex there are Agena-qualified umbilical connections for propellant loading (UDMH/IRFNA), propellant venting and scavenging, high pressure gas loading, electric power, and air conditioning. The Agena D's J-100 electrical umbilical plug has 26 No. 16AWG circuits reserved for spacecraft use.

The umbilical connectors fall into the following three types:

a. Lanyard disconnect for quick disconnect at lift-off
b. Pneumatic Release
c. Test/service plugs to be removed at the time of gantry removal

The lanyard types may be actuated by one of the following three modes:

a. Retraction module cylinder lanyard pull
b. Flag or boom retraction lanyard pull
c. Flyaway lanyard pull

A backup system of flyaway safe shear is designed into some of the connections. This system shears the dogs on the ground half of the connector at a load level that will not harm the spacecraft, should the airborne half fail to disengage promptly.

The Agena flight qualified umbilicals listed in Table 18-1 can be used for servicing the spacecraft up to lift-off.

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Table 18-1

AGENA UMBILICAL DATA

Mechanical Connectors	Working Pressure (psig)	Flow Rate
Propellant Fills (UDMH/IRFNA)	100	1-1/2-in. diam. equivalent flow
Propellant Vents	100	20 scfm of H_e at 30 psig and $70^{\circ}F$, $\Delta p <$ 1/2 psi
Gas Fill (N_2/H_e)	3600	40 scfm of N_2 at 1000 psig, $\Delta p < 100$ psi
Air Conditioning	1	75 lb/min at 1 psig
Electrical Connectors	**No. Connectors**	**Wire Size and Coaxial Type**
J-100 (Main Agena Electrical)	108	No. 16 AWG
	2	No. 4 AWG
	4	RF Coaxial for RG – 142/U
Cole (Mariner-C Spacecraft)*	69	No. 20 AWG
	2	RF Coaxial for RG – 142/U
J-26	24	No. 20 AWG

*Also measures spacecraft environment temperature.

Orientation of the listed umbilical connectors relative to the launch pad umbilical boom is shown in Figure 18-13.

The primary signal used for umbilical disconnect is RVOC* for Atlas launches and MFVO** for TAT launches. The other signals used are:

(1) Flight lock-in signal for Atlas backup

(2) Lift-off for TAT and Atlas backup

Test plugs and service connections that can be removed two hours before lift-off need not be the quick disconnect type and will be removed by hand before the time of gantry removal.

*Release Valve Off Charge

**Main Fuel Valve Open

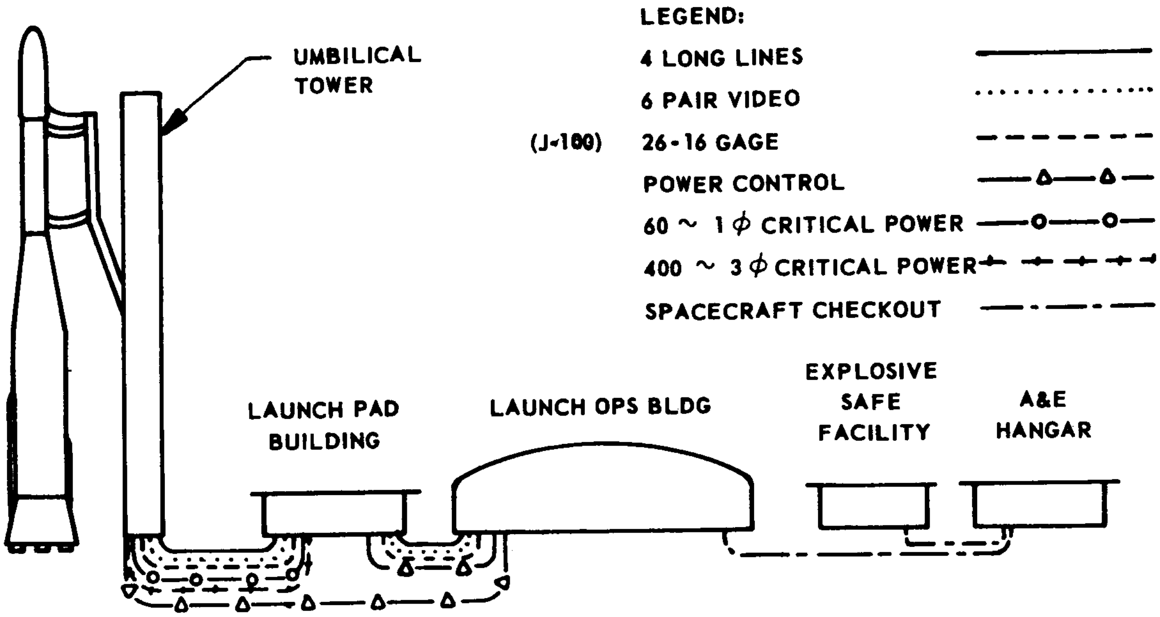

Figure 18-5 Launch Complex 12 and 13 Wiring

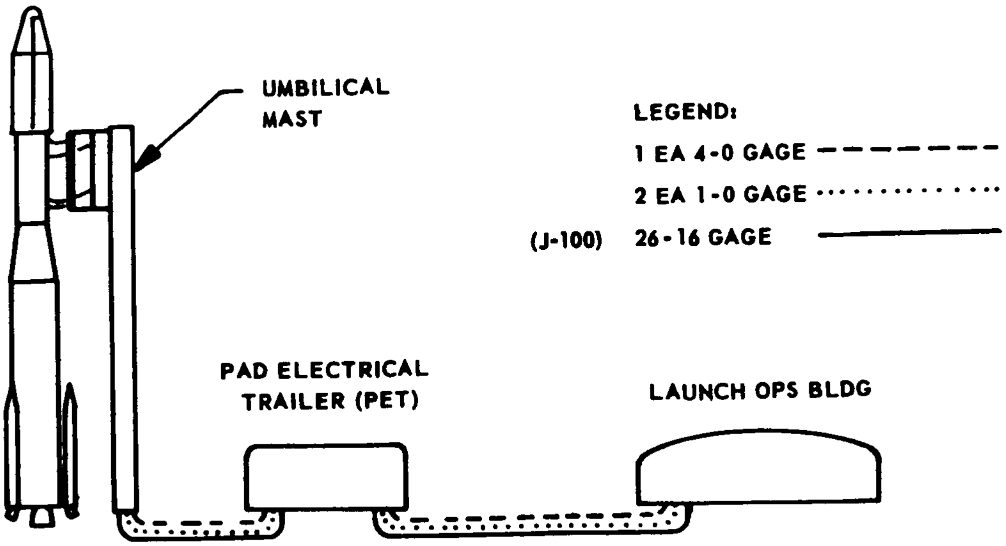

Figure 18-6 Launch Complex 75-1-1 Wiring

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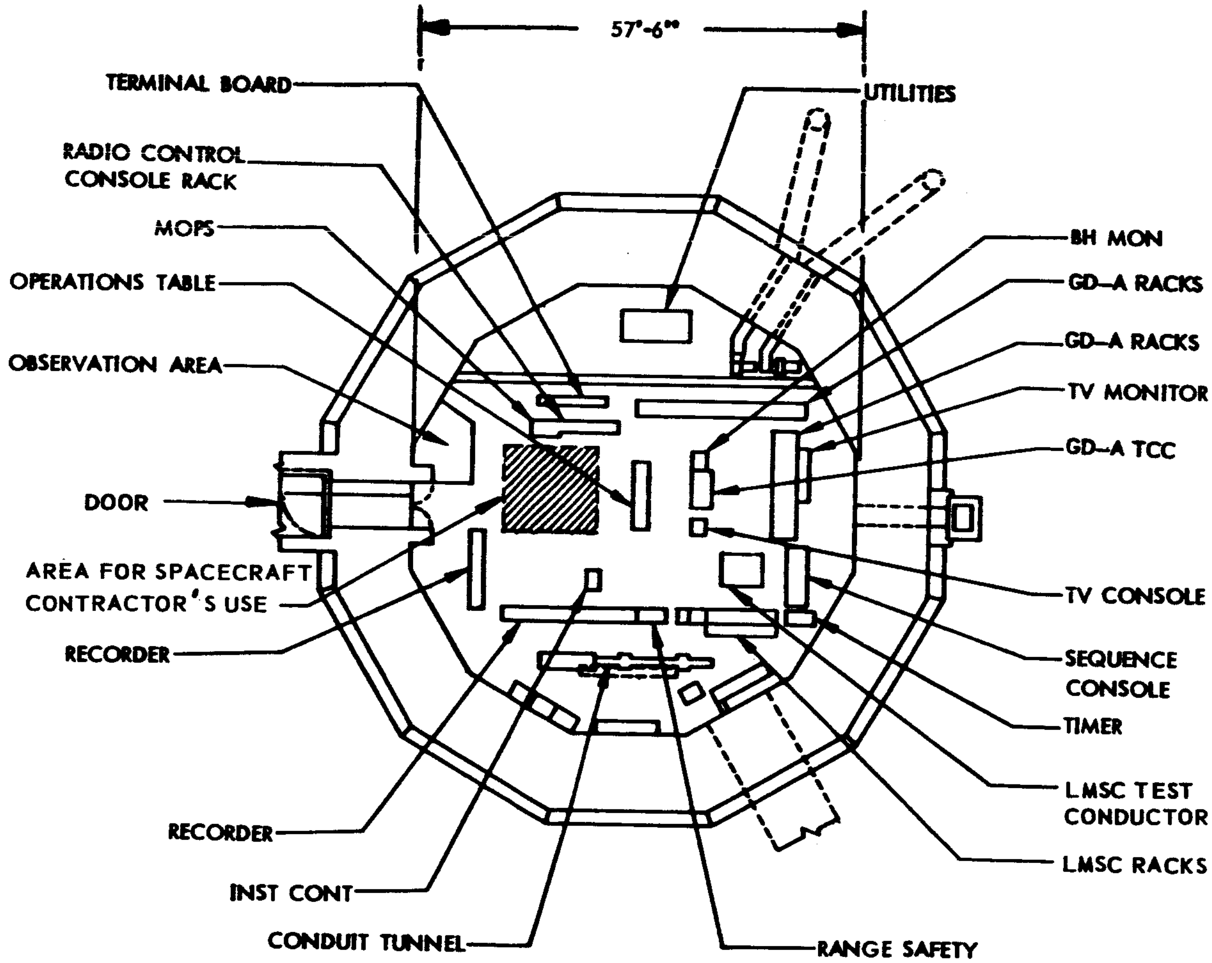

Figure 18-7 Launch Operations Building (Blockhouse) Launch Complex 12

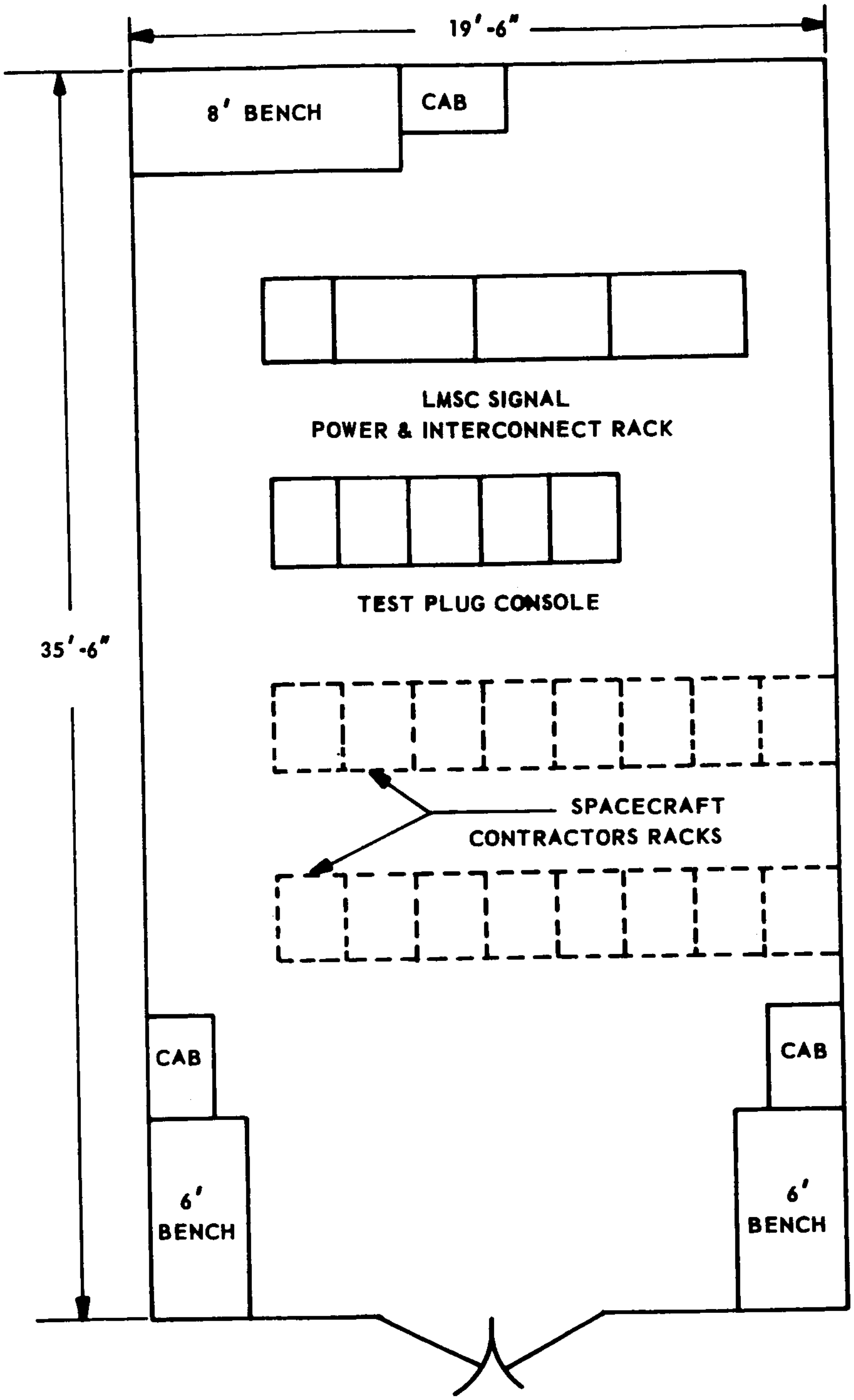

Figure 18-8 Launch Pad Building – Launch Complex 12

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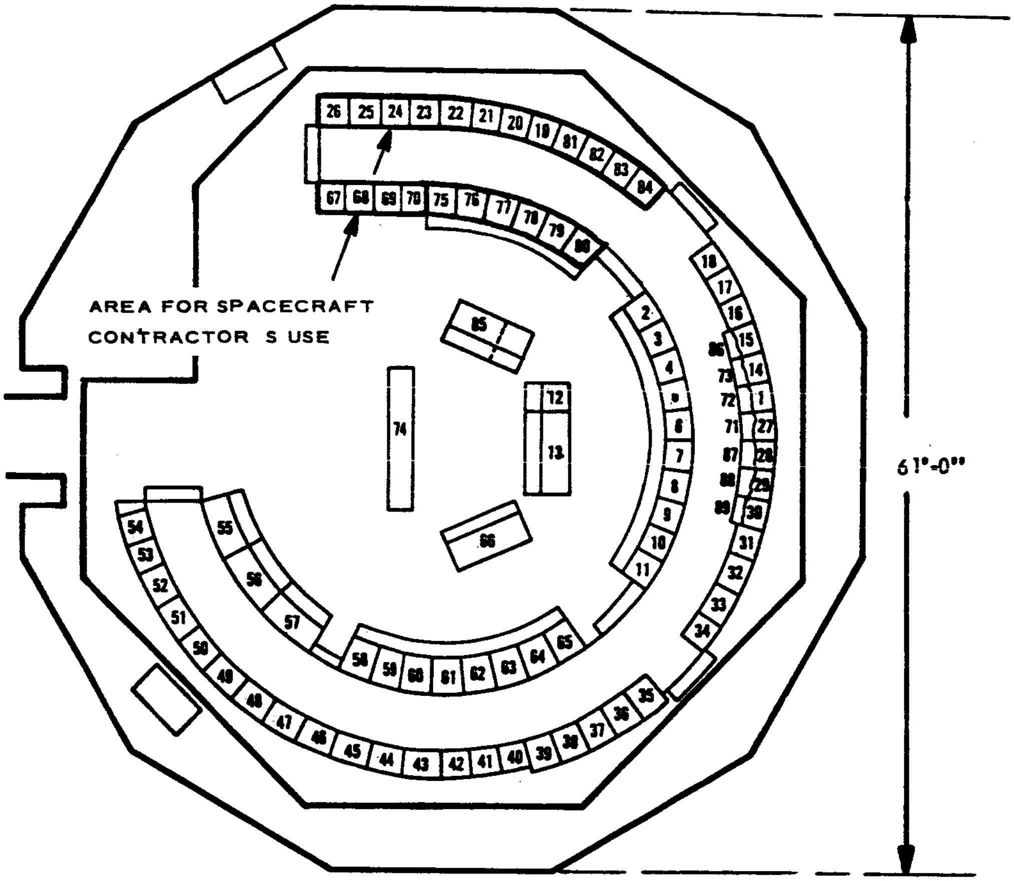

LEGEND

LOCATION NO.	EQUIPMENT	LOCATION NO.	EQUIPMENT
1	AUXILIARY, MSL, AND FACILITY POWER	37	AGENA SEQUENCER RECORDER
2	AUTOPILOT CONSOLE NO. 1	38 - 40	CEC TAPE SYSTEM
3	AUTOPILOT CONSOLE NO. 2	41	EQUIPMENT STORAGE RACK
4	GUIDANCE CONSOLE	42	ANALOG PATCH PANEL
5	HOLDDOWN/RELEASE AND HYDRAULIC CONSOLE	43 - 50	BROWN GRAPHIC RECORDER RACK
6	FUEL CONSOLE	51	SPARE RACK
7	LO_2 TANKING CONSOLE	52 - 54	OSCILLOGRAPH RECORDER RACK
8	PNEUMATIC CONSOLE	55	SPACE
9	TELEMETRY AND RANGE SAFETY CONSOLE	56	TV CONTROL CONSOLE
10	WATER SYSTEM CONSOLE	57	INSTRUMENTATION CONTROL CONSOLE
11	MISSILE POWER CONSOLE	58	AGENA COMMUNICATIONS CONSOLE
12	BLOCKHOUSE MONITOR CONSOLE	59 - 61	AGENA ELECTRICAL & GUIDANCE CONSOLE
13	TEST CONDUCTOR'S CONSOLE	62, 63	AGENA PNEUMATIC & PROPULSION CONSOLE
14	SPARE RACK	64	AGENA POWER SUPPLY
15 - 17	SANBORN RECORDER RACK	65	AGENA INTERCONNECT BOX
18	AUTOPILOT RACK NO. 1	66	AGENA LAUNCH CONDUCTOR'S CONSOLE
19, 20	SPACE	67 - 69	SPACE
21 - 25	BROWN GRAPHIC RECORDER RACK	70	TEST COORDINATORS CONSOLE
26	SPARE RACK	71 - 73	TV MONITOR
27	RACK ASSEMBLY ENGINE TEST, AND PURGE	74	COMMUNICATIONS & OPERATIONS TABLE
28	RANGE SAFETY COMMAND CHECKOUT SET	75	JUNCTION BOX
29, 30	SPARE RACK	76	GUIDANCE AND CONTROL
31	EQUIPMENT STORAGE RACK	77 - 79	DATA ENCODER
32	SEQUENCE RECORDER & PATCH PANEL	80	SPACECRAFT RF POWER MONITOR CONSOLE
33	SEQUENCE RECORDER RACK	81 - 84	SPACE
34	SOUND RECORDER RACK	85	FSV MASTER LCC
35, 36	AGENA SANBORN RECORDER RACK	86	PROVISIONS FOR TV MONITOR
		87 - 89	COUNTDOWN CLOCKS

Figure 18-9 LOB — Launch Complex 13

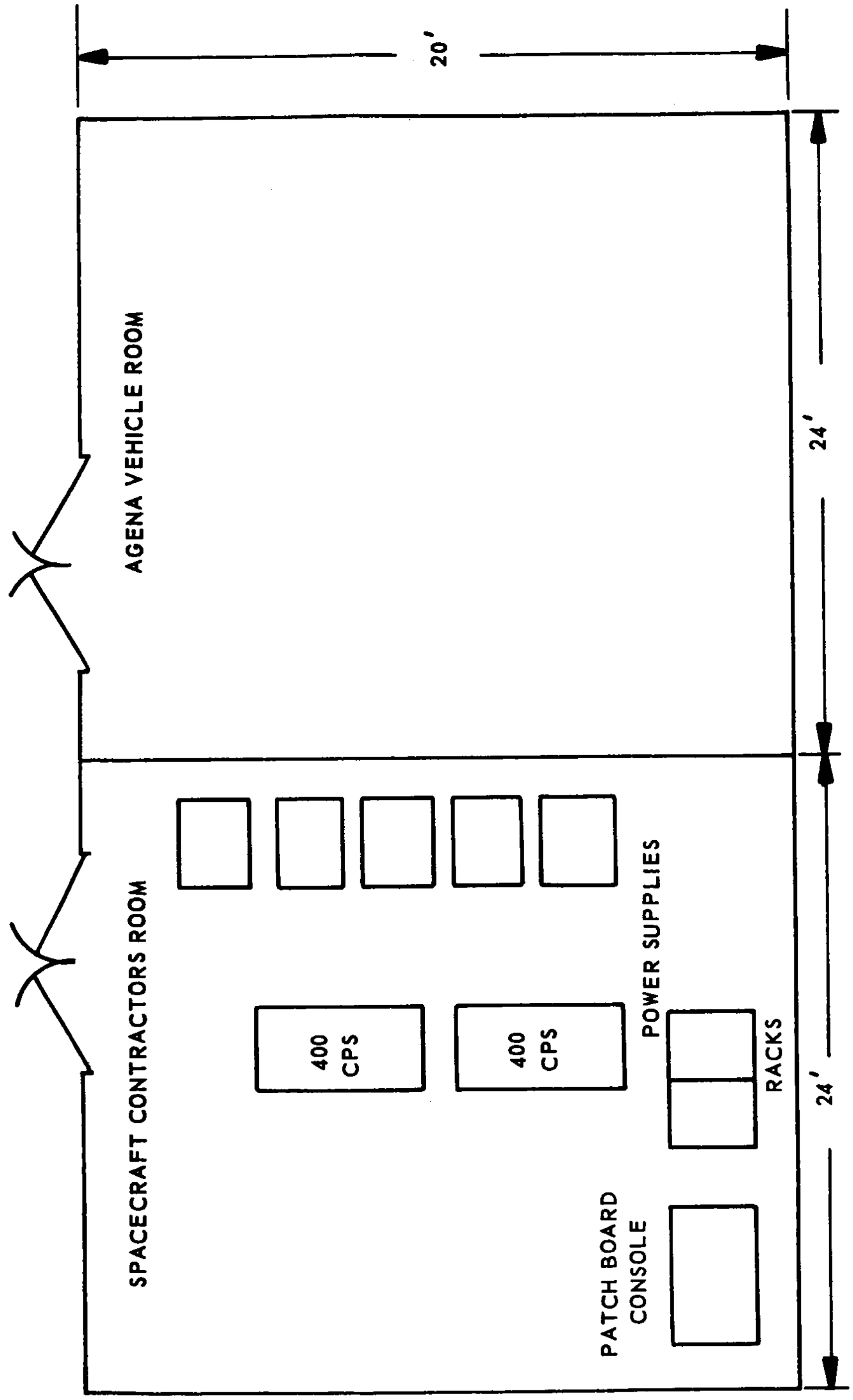

Figure 18-10 Launch Pad Building – Launch Complex 13

E-3236-5

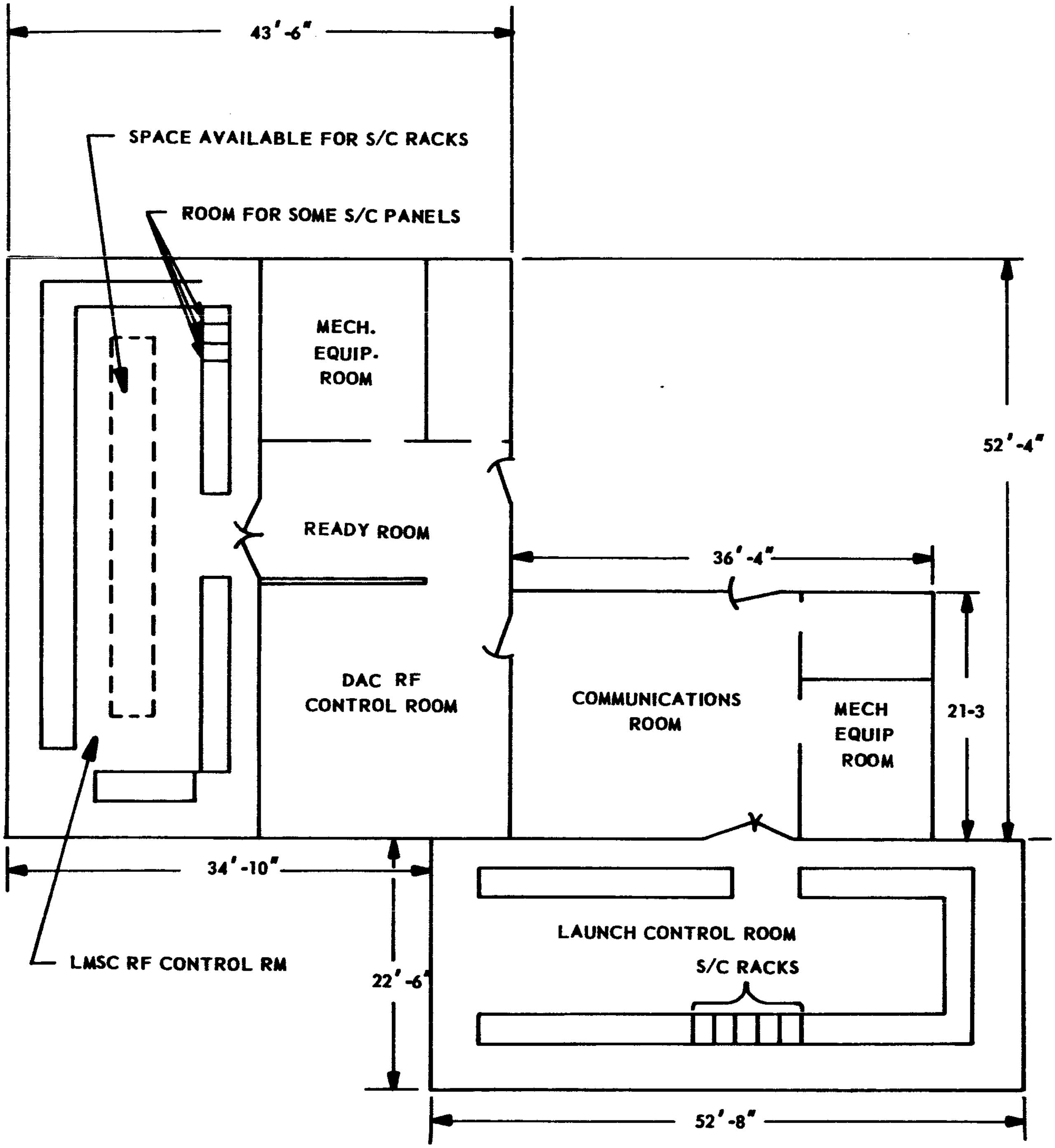

Figure 18-11 Launch Operations Building – Complex 75-1-1

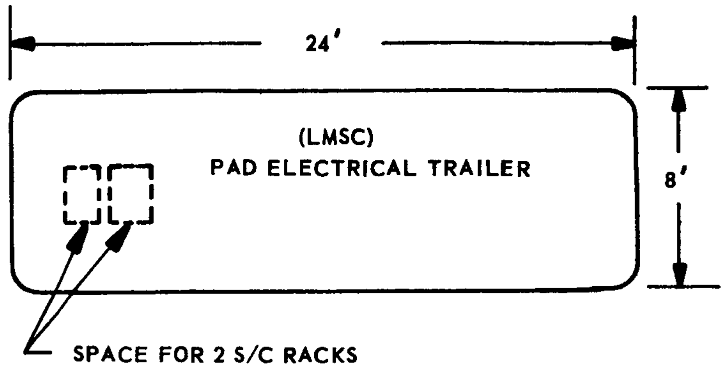

Figure 18-12 Launch Complex 75-1-1, Pad Electrical Trailer

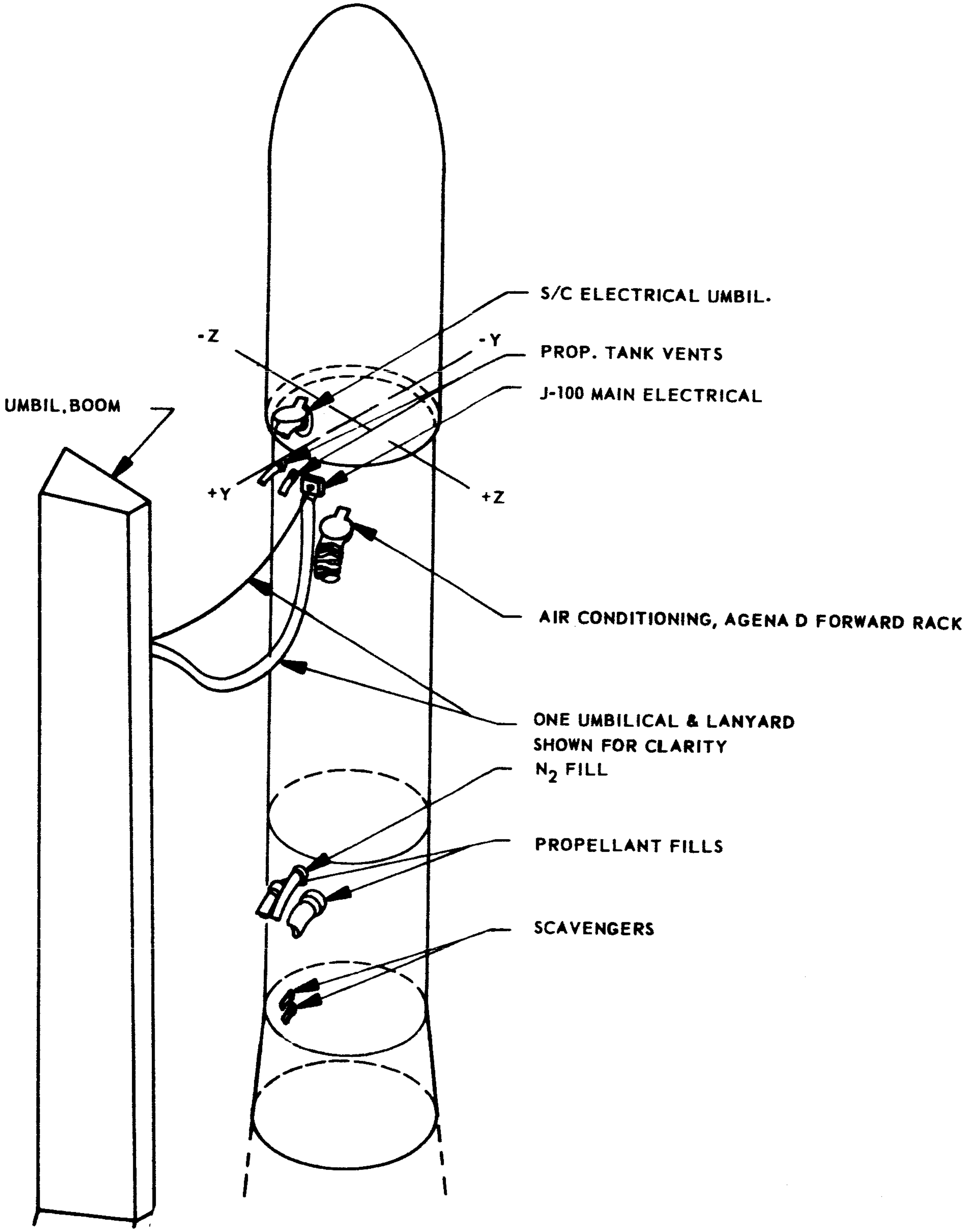

Figure 18-13 Umbilical Orientation

18.4.2 Existing Launch Pad Air Conditioning Equipment for Agena and Spacecraft

Portable and semi-portable air conditioning equipment exists on the various launch complexes to provide conditioned air for internal and external shroud cooling. The spacecraft environments that can be provided are discussed in Section 8. The available air conditioning equipment is listed below:

Launch Complex	Air Conditioning Equipment
LC 75-1-1	One VP 45 with one Type 15A as required
LC 12	Five Type 15A
LC 13	One VP 45 and one Type 15A

VP 45 output for spacecraft is 70 lb/min ±3 lb/min, 25°F to 80°F, and 40 per cent relative humidity.

Type 15A is 50 to 150 lb/min ±5 lb/min, 30°F to 100°F, and 100 to 15 per cent relative humidity.

18.5 EXTERNAL TYPE SHROUD COOLING EQUIPMENT

On-pad cooling for the Ranger and Mariner D spacecraft is provided by external cooling blankets which function as heat exchangers. Conditioned air from the launch pad air conditioning units is ducted into the wrap-around fabric blanket or jacket that is secured to the shroud with a parachute type quick release cable. Construction is such that the cooling air is introduced through a manifold entrance duct to semi-circular ducts carrying the air around the shroud. Vents at the ends of these ducts exhaust the air at a predetermined rate. Umbilical boom retraction and fly-away modes provide for cooling blanket release and retraction. Figure 18-14 illustrates the Ranger and Mariner D type cooling blankets.

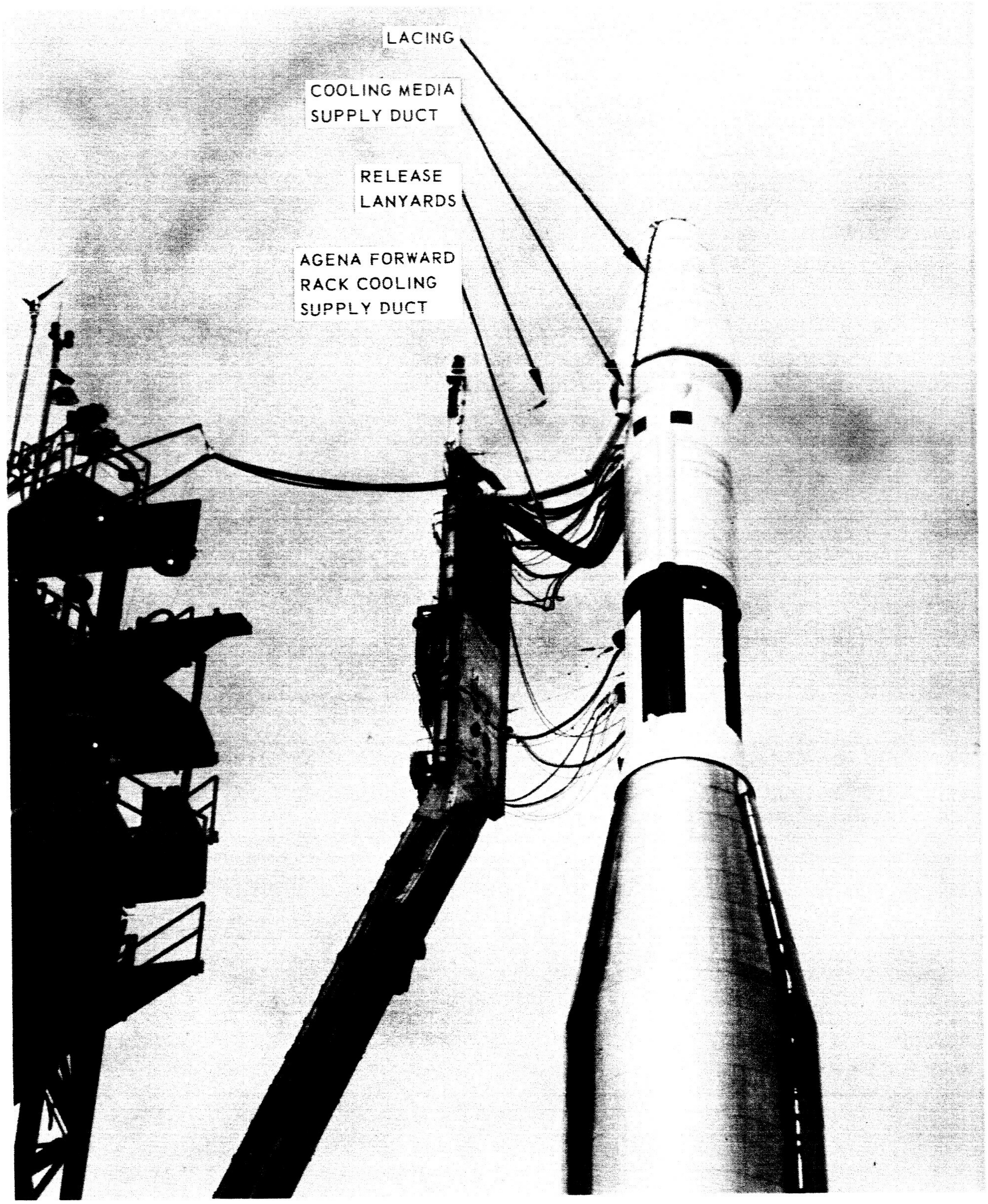

Fig. 18-14 Ranger-Mariner D Type Temperature Control Blanket

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The external cooling equipment is summarized in Table 18-2

Table 18-2
EXTERNAL SHROUD COOLING EQUIPMENT

LMSC P/N	Shroud Type	Nomenclature	Function
1460699-1	Mariner	Blanket, Temperature Control	Used with air conditioning unit to provide proper temp. environment for S/C
1460023	Ranger	Blanket, Temperature Control, Type 5	Same

18.6 SPACECRAFT TRANSPORTATION AND HANDLING EQUIPMENT

Ground handling equipment (GHE) is used for handling and supporting the spacecraft while being assembled, disassembled, transported, transferred, stored, lifted, tested, aligned and mated with the Agena vehicle. Certain GHE may be supplied by LMSC for particular spacecraft handling usage. A number of GHE items are provided for handling the Mariner D and Ranger spacecraft and adapters during encapsulation within the shroud. Figures 18-15 through 18-17 illustrate the manner of encapsulation and subsequent handling of the Mariner D spacecraft. Figure 18-18 illustrates the Ranger type shroud/spacecraft hoisting equipment. Tables 18-3 and 18-4 summarize the Mariner D and Ranger shroud/spacecraft handling equipment and provide functional descriptions for each major item.

Table 18-3

MARINER D TYPE SHROUD/SPACECRAFT HANDLING EQUIPMENT

LMSC/P/N	Nomenclature	Function
JPL-9212 (JPL P/N)	Transporter, spacecraft assembly	Towed trailer used for transporting spacecraft within Hangar A & E, and between hangar and pad when encapsulated in shroud.
1546463	Fixture, adapter/spacecraft support ring	Used to support the transition/spacecraft adapter and spacecraft when mating shroud to spacecraft transporter.
1548645	Tool, spacecraft ejection spring cocking	Used to compress spacecraft ejection springs during mating.
1547574	Tool, spacecraft ejection spring locking	Used to hold spacecraft ejection springs compressued during mating.
1544996	Ring adapter, spacecraft assembly hoisting	Structural ring for installation on shroud to provide pickup points for hoisting encapsulated spacecraft.
1544997	Sling, spacecraft assembly hoisting.	Used with P/N 1544996 to hoist encapsulated spacecraft assembly for Agena/payload mating.
1544654	Container, adapter/spacecraft fixture ring	Used for shipping of fixture P/N 1546463.
1548720	Cover, spacecraft V-band protective	Molded rubber V-band cover to provide personnel protection when V-bands are installed.
1548818	Sling, payload safety	Used with P/N 1544997 when hoisting encapsulated spacecraft assembly. Provides means of arresting spacecraft assembly in the event of a shroud V-band failure.

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Table 18-4

RANGER TYPE SHROUD/SPACECRAFT HANDLING EQUIPMENT

LMSC P/N	Nomenclature	Function
1586648	Sling, vertical hoisting	2000# capacity sling for hoisting shroud/spacecraft assembly for mating.
1586649	Yoke assembly, handling	Assembly for attaching to shroud/ spacecraft assembly to provide pickup points for sling, P/N 1586648.
1532301	Cover, protective payload adapter	Glass fabric cover used to seal bottom of shroud/spacecraft assembly during hoisting and transporting.
JPL P/N	Transporter, spacecraft assembly	Towed trailer for transporting spacecraft assembly within Hangar A & E, and between hangar and launch pad.

18.7 SHROUD HANDLING EQUIPMENT

The shrouds discussed in Section 10 are of three basic configurations as regards to handling equipment; these are the Nimbus type, the Ranger type and the Mariner D type. The Nimbus shroud handling equipment listed in Table 18-5 is also utilized on the S-27, Comsat, and EGO shrouds with only slight variations necessary for use on the POGO and Gemini shrouds.

The Ranger shroud handling equipment is listed in Table 18-6 along with a functional description of each major item. The Ranger shroud handling sling and Ranger shroud handling dolly are illustrated in Figures 18-19 and 18-20. Table 18-7 summarizes the Mariner D shroud handling equipment. Because of the small clearance between the Mariner D shroud and spacecraft a shroud guide fixture is needed during the spacecraft encapsulation operation. Application of this item of equipment (P/N 1545752) is illustrated in Figure 18-15 along with the shroud handling sling. The guide rails attached to the adapter and spacecraft support fixture are engaged by the shroud fixture rails to insure that the shroud does not strike the spacecraft during encapsulation.

Table 18-5

NIMBUS TYPE SHROUD HANDLING EQUIPMENT

LMSC P/N	Nomenclature	Function
1460520	Strap, interim holding	To hold shroud halves mated while installing flight bands.
1460521	Trailer, shroud handling	For storing and transportation of shroud between MAB and pad.
1460522	Cover, shroud	To protect shroud while in trailer.
1460524	Fixture, shroud installation	Utilized for mating or demating shroud to Agena.
1460525	Hoisting adapter, shroud	Provides attach points for attaching hoisting sling when removing shroud from trailer and installing on installation fixture.

Table 18-6

RANGER TYPE SHROUD HANDLING EQUIPMENT

LMSC P/N	Nomenclature	Function
1586650	Sling, shroud handling	Used for hoisting shroud when mating shroud to spacecraft support structure.
1586646	Dolly, shroud handling	Used to stow and transport shroud within Hangar A & E.
1591502 or 1518967	Tool, spring cocking	Used to compress separation springs in shroud prior to mating.
1591503	Tool, spring locking	Used to lock separation springs in compressed position.

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Table 18-7

MARINER-D TYPE SHROUD HANDLING EQUIPMENT

LMSC P/N	Nomenclature	Function
1546943	Sling, shroud handling	Used for hoisting shroud when mating shroud to vehicle transition section.
1546785	Tools, spring cocking and locking	Used to compress and secure separation springs in shroud prior to mating.
1545569	Dolly, shroud handling	Used to stow and transport shroud within Hangar A & E.
1545752	Fixture, shroud guide	Used for mating shroud to spacecraft assembly.
1546945	Cover, shroud protection	Used to cover shroud when installed in shroud dolly.
1546841	Cover, shroud ejection spring protective	Used to protect ejection springs when installed in shroud.
1547799	Cover, shroud V-band protective	Used to provide personnel protection when V-bands are installed on shroud.

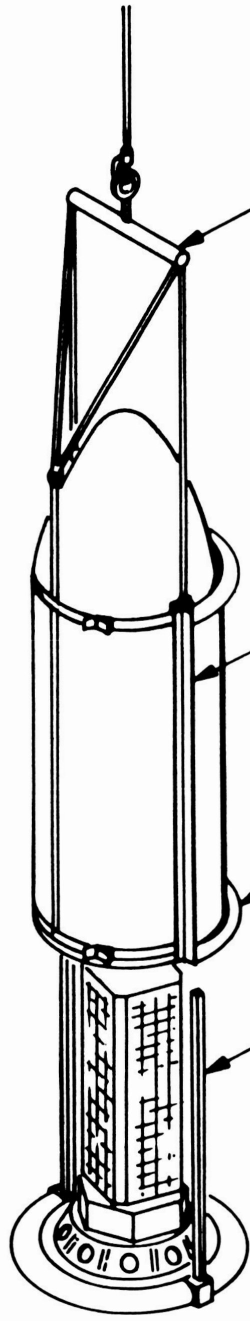


Fig. 18-15 Mariner D Shroud Installation Fixture

E-3236-5

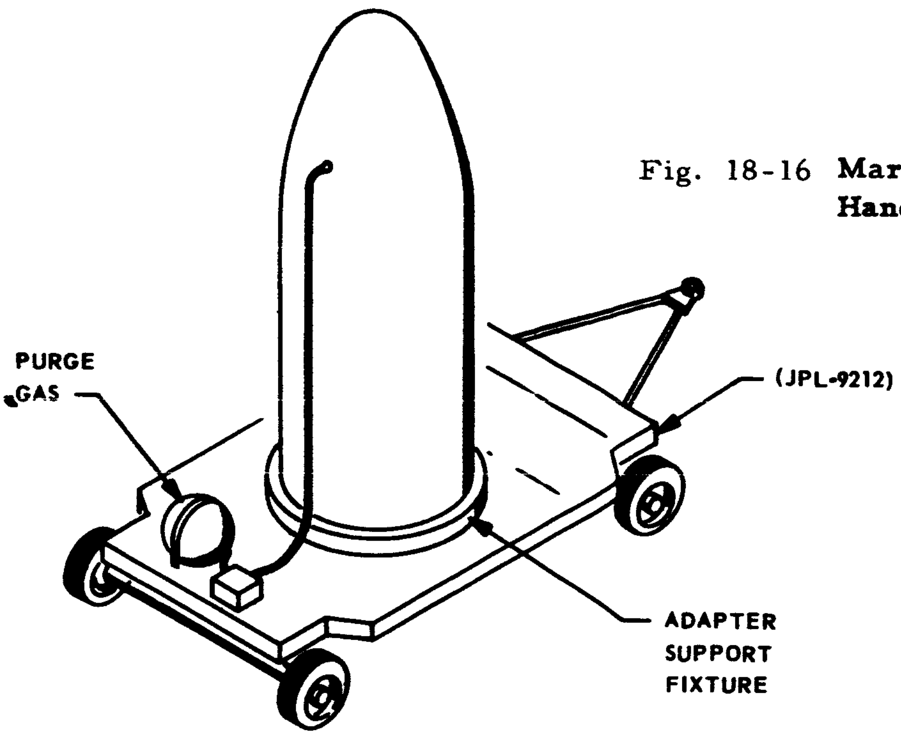


Fig. 18-16 **Mariner D Spacecraft Handling Dolly**

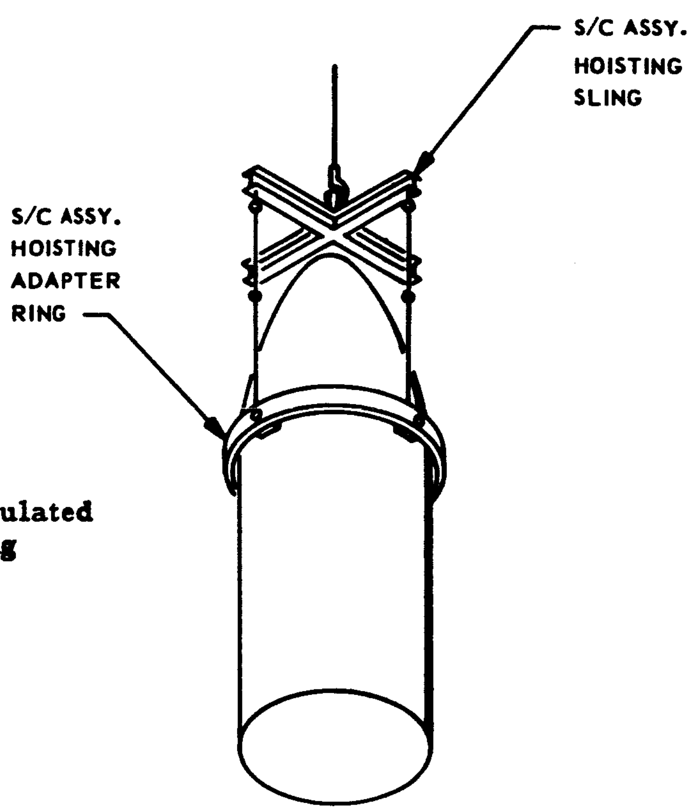
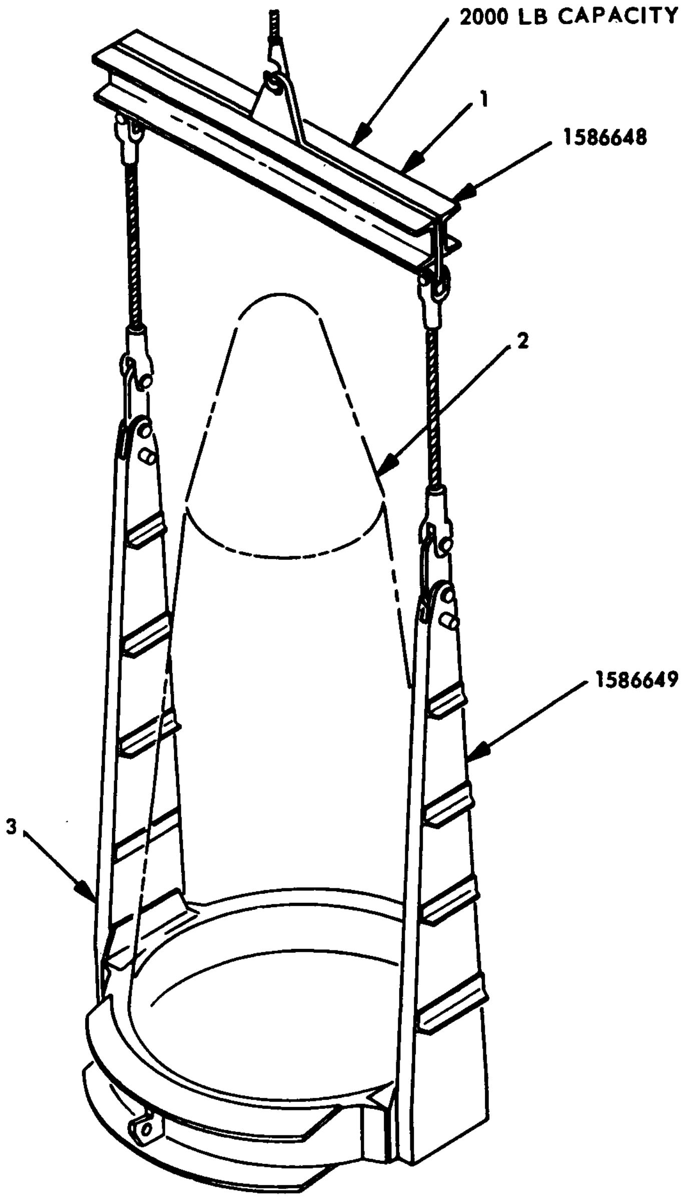


Fig. 18-17 **Mariner D Encapsulated Spacecraft Hoisting**



1. NASA SPACECRAFT VERTICAL HOISTING SLING
2. SHROUD (REF)
3. HANDLING YOKE ASSEMBLY BEAM ASSEMBLY (2 REQD)

Figure 18-18 Ranger Type Shroud/Spacecraft Hoisting Sling Assembly

E-3236-5

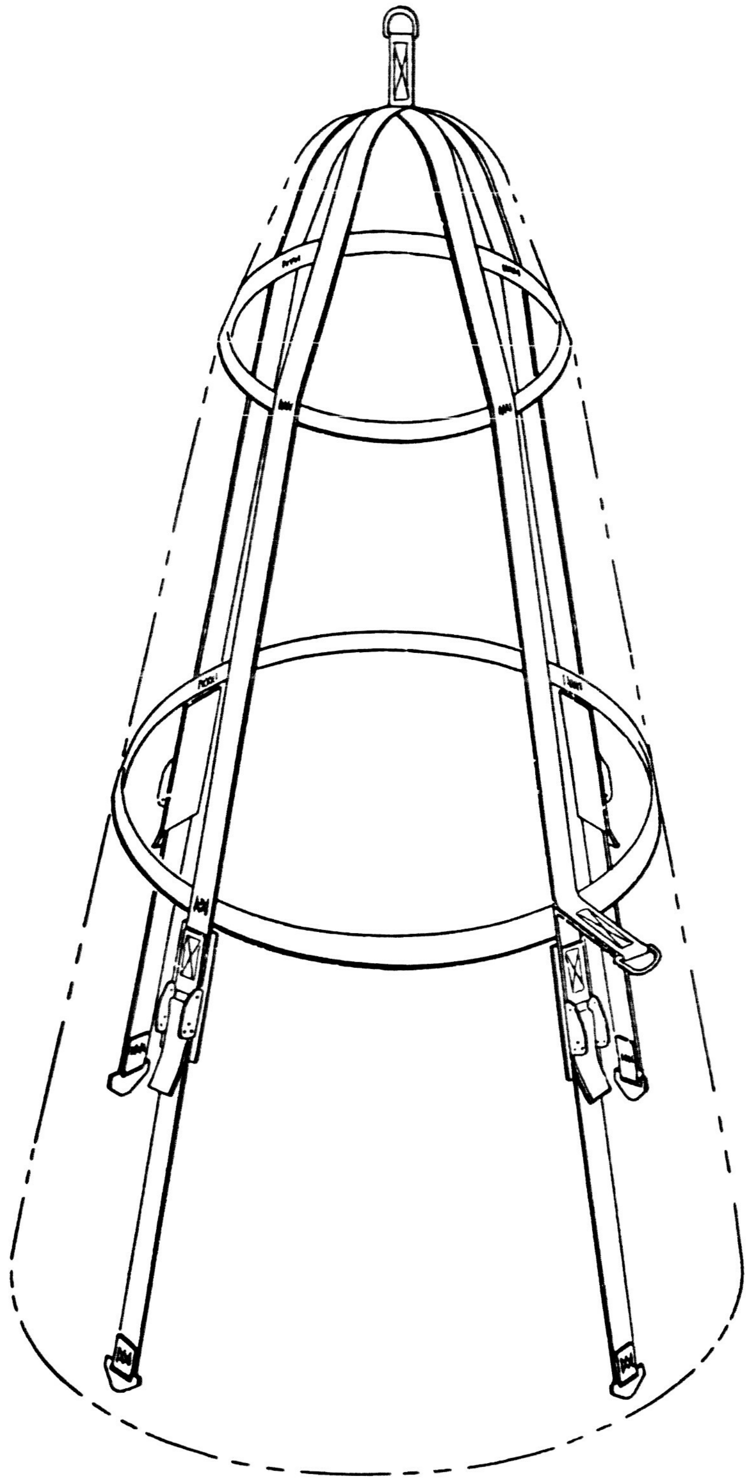


Figure 18-19 Ranger Shroud Handling Sling, P/N 1586650

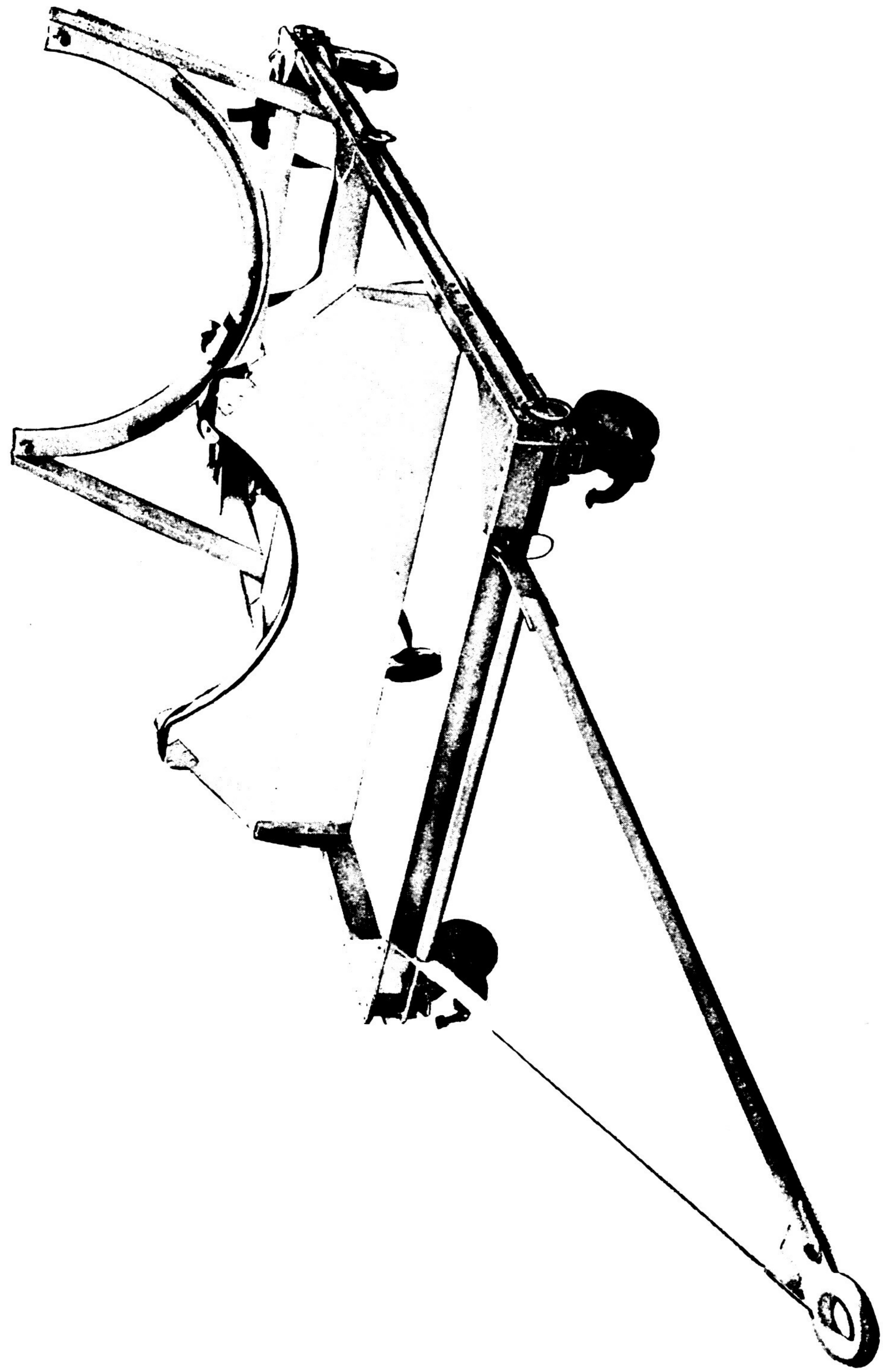


Figure 18-20 **Ranger** Shroud Handling Dolly, P/N 1586646

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SECTION 19
PRELAUNCH AND LAUNCH ACTIVITIES

19.1 GENERAL

This section discusses the major prelaunch activities, tests, and launch operations necessary for the Atlas/Agena/spacecraft and the TAT/Agena/spacecraft vehicle configurations. These activities include the prelaunch test sequence, range countdown, Agena on-stand hold capability, launch restraints, and range support for each configuration. The test sequences are based on the philosophy of minimum testing requirements consistent with high reliability. A typical spacecraft checkout sequence is also presented in order to indicate how it folds into booster vehicle operations. Combined tests in which the spacecraft or simulator is required to participate are also briefly described.

19.2 PRELAUNCH TEST SEQUENCE

19.2.1 Atlas/Agena (ETR)

A detailed launch operations flow chart for the Atlas/Agena/spacecraft vehicle combination is presented in Fig. 19-1. Time sequencing of the major checkout milestones is presented in Fig. 19-2. In the past, NASA programs have launched the Atlas/Agena combination from ETR exclusively; and, therefore the test terminology and philosophy is slanted in that direction. However, the basic ideas may be carried over for an Atlas/Agena WTR launch.

Missile Assembly Building (MAB) operations conducted on the Atlas are limited to quality control type inspection. The vehicle is then transported to the launch complex and erected on the launch stand where detailed systems and component checkout is performed.

The Agena is completely checked out at the MAB. All subsystems are functionally tested and validated with critical time-limited Agena system validations performed prior to transport to the launch complex. It is recommended that the final spacecraft or spacecraft simulator mechanical fit and electrical checks, and the shroud fit check, be conducted in the LMSC MAB during this period in order to avoid delays at the launch complex.

The booster adapter is mated to the Atlas several days prior to the Agena arrival at the launch complex. The Agena is transported to the launch stand, and inaccessible pyrotechnics are installed prior to mating. As indicated in Fig. 19-1 compatibility tests are performed between the Agena subsystems and the AGE. The spacecraft or simulator is also mated during this time and launch complex compatibility tests accomplished.

The spacecraft shroud or other ground protective covering is installed at this time and is removed and reinstalled as required.

The following combined operations are performed on the space vehicle (Atlas/Agena/spacecraft) to verify flight readiness:

a. A combined radio frequency interference test is normally conducted on the complete flight vehicle. This test verifies the compatibility of the space vehicle and ground RF systems. Noninterference of the RF and electromagnetic signals with the vehicle pyrotechnic circuits is also verified.

b. The J-FACT (Joint Flight Acceptance Composite Test), consisting of a launch countdown operation and ascent flight simulation, is conducted to demonstrate and evaluate the electronic system readiness of the complete space vehicle and ground support systems.

c. A full dress rehearsal (simulated launch) of the launch countdown is conducted on the complete space vehicle to verify complete systems readiness prior to initiation of the launch countdown.

Following the combined vehicle tests, the final (F-day) flight hardware installations and preflight checks are made on the Atlas/Agena D/spacecraft.

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ATLAS

1. RECEIVING AND INSPECTION
2. PROPELLANT UTILIZATION SYSTEM CHECKOUT AND CALIBRATION
3. PROGRAMMER EVALUATION
4. GYRO TEST
5. EVALUATION OF FILTER SERVO CAN

AGENA

1. RECEIVING & INSPECTION
2. VEHICLE BUS RESISTANCE CHECK
3. HORIZON SENSOR ALIGNMENT AND HEAD PRESSURE TEST
4. BEACON SYSTEM VALIDATION TEST
5. VEHICLE INSTRUMENTATION CALIB
6. DESTRUCT SYSTEM TEST
7. TELEMETRY SYSTEM VALIDATION
8. GUIDANCE SYSTEM VALIDATION AND FINAL ADJ.
9. GUIDANCE AND FLIGHT CONTROL SYSTEM PRE-FLIGHT CHECKS
10. PROPULSION SYSTEM LEAK AND FUNCTIONAL TESTS AND ENGINE SERVICING
11. VEHICLE WEIGHT DETERMINATION

---

12. AGENA/SPACECRAFT ELECTRICAL INTERFACE CONTINUITY TEST
13. AGENA/SPACECRAFT/SHROUD ALIGNMENT TEST

SPACECRAFT OR INTERSTAGE

SPACECRAFT

1. RECEIVING AND INSPECTION
2. SUBSYSTFM VALIDATION CHECKS & COMBINED SYSTEMS RUN
3. EQUIPMENT INSTALLATIONS

ATLAS

ERECT VEHICLE

1. PROPULSION SYSTEM LEAK & FUNCTIONAL CK ENGINE ALIGNMENT
2. PROPELLANT UTILIZATION SYSTEM LEAK CK
3. TELEMETRY SYSTEM BLOCKHOUSE COMPATIBILITY AND FUNCTIONAL TESTS
4. RELEASE SEQUENCE TEST
5. MISSILE ELECTRICAL BLOCKHOUSE COMPATIBILITY TEST
6. RANGE SAFETY COMMAND SYSTEM CHECKOUT
7. MODE IIIG BLOCKHOUSE & AIRBORNE GUIDANCE CK
8. GUIDANCE COMMAND TEST
9. AUTOPILOT SQUIB CIRCUITRY CK
10. INTEGRATED LAUNCH CONTROL TEST
11. BOOSTER TANKING TEST
12. BOOSTER FLIGHT ACCEPTANCE COMPOSITE TEST
13. MATE BOOSTER ADAPTER

AGENA

1. MATING PROCEDURE
2. PYROTECHNIC INSTALLATION
3. PROPULSION SYSTEM PRESSURE TEST AND INSPECTION
4. ELECTRICAL SYSTEM/AGE COMPATIBILITY TESTS
5. GUIDANCE SYSTEM/AGE COMPATIBILITY TESTS
6. TELEMETER & BEACON RF FUNCTIONAL TESTS

SPACECRAFT

1. MATING PROCEDURE
2. SPACECRAFT/AGE COMPATIBILITY TESTS (BLOCKHOUSE COMPATIBILITY CHECKS) RF READOUT
3. SHROUD MATING

1. COM
INTE
2. JOIN
COM
3. SIMU

MISSILE ASSEMBLY BUILDING TESTS

LAUNCH C

1

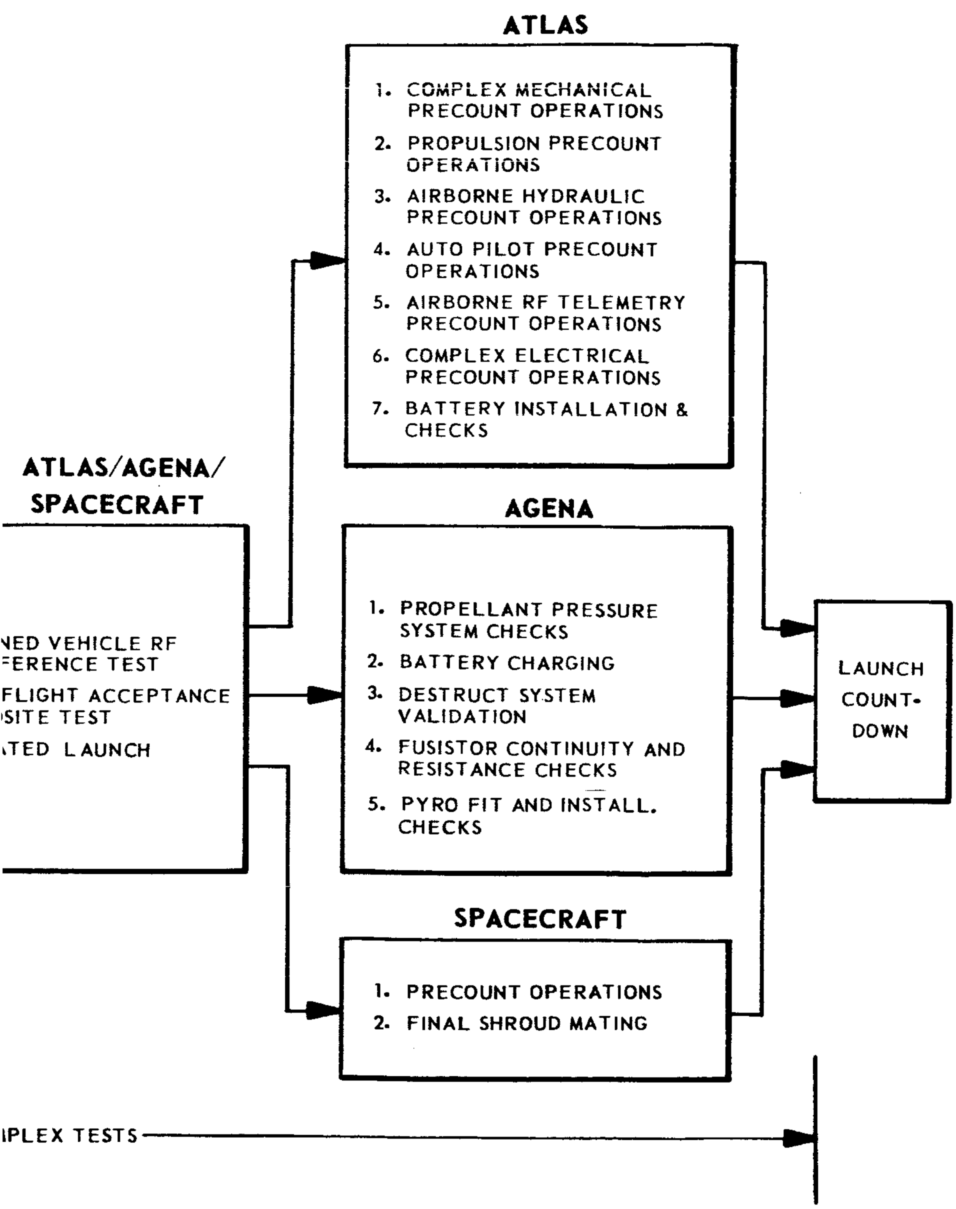

Figure 19-1 Typical Atlas/Agena D/Spacecraft Prelaunch Operations and Test Sequences

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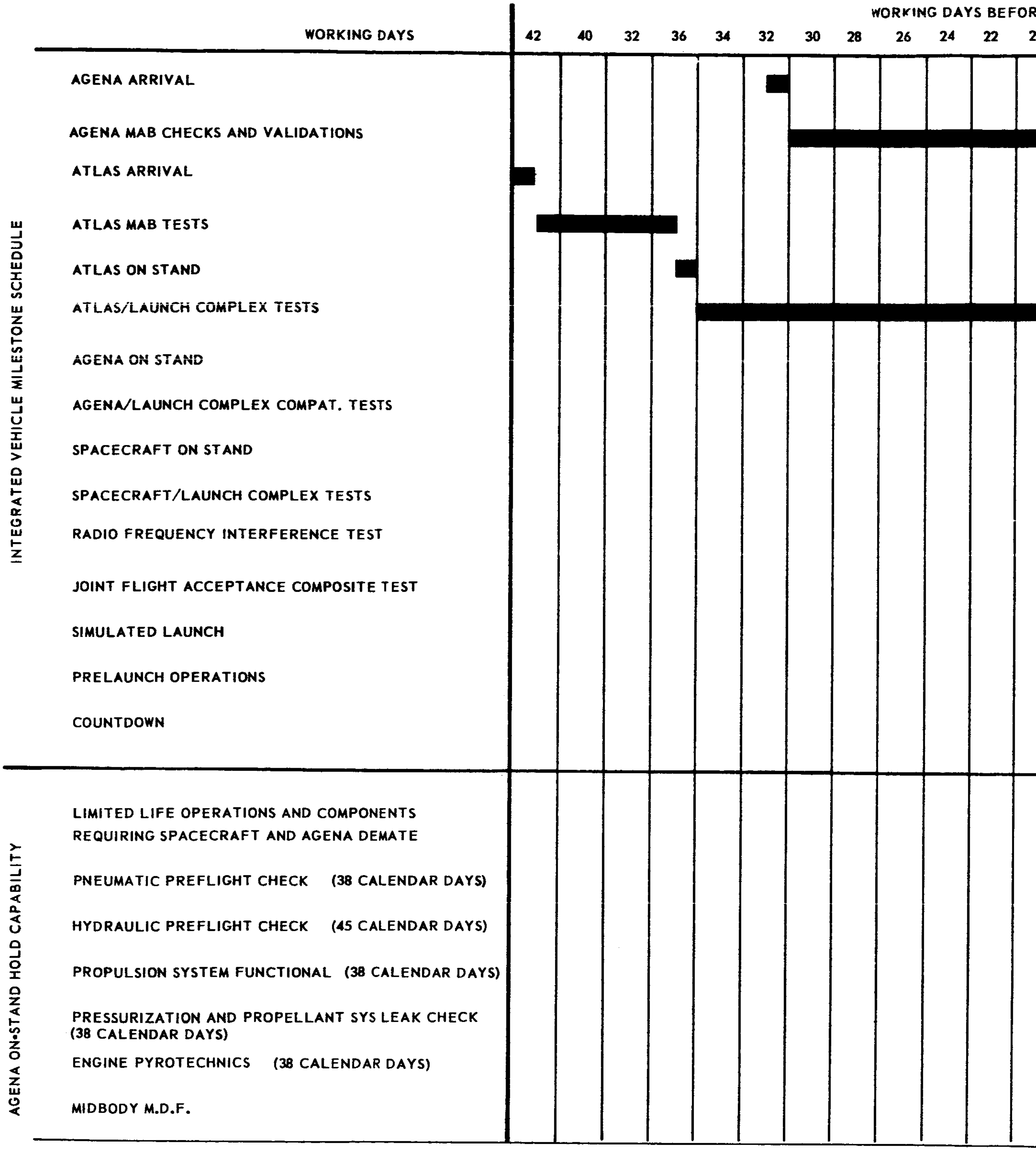


WORKING DAYS BEFOR
WORKING DAYS
42 40 32 36 34 32 30 28 26 24 22 2
INTEGRATED VEHICLE MILESTONE SCHEDULE
AGENA ARRIVAL
AGENA MAB CHECKS AND VALIDATIONS
ATLAS ARRIVAL
ATLAS MAB TESTS
ATLAS ON STAND
ATLAS/LAUNCH COMPLEX TESTS
AGENA ON STAND
AGENA/LAUNCH COMPLEX COMPAT. TESTS
SPACECRAFT ON STAND
SPACECRAFT/LAUNCH COMPLEX TESTS
RADIO FREQUENCY INTERFERENCE TEST
JOINT FLIGHT ACCEPTANCE COMPOSITE TEST
SIMULATED LAUNCH
PRELAUNCH OPERATIONS
COUNTDOWN
AGENA ON-STAND HOLD CAPABILITY
LIMITED LIFE OPERATIONS AND COMPONENTS REQUIRING SPACECRAFT AND AGENA DEMATE
PNEUMATIC PREFLIGHT CHECK (38 CALENDAR DAYS)
HYDRAULIC PREFLIGHT CHECK (45 CALENDAR DAYS)
PROPULSION SYSTEM FUNCTIONAL (38 CALENDAR DAYS)
PRESSURIZATION AND PROPELLANT SYS LEAK CHECK (38 CALENDAR DAYS)
ENGINE PYROTECHNICS (38 CALENDAR DAYS)
MIDBODY M.D.F.

1

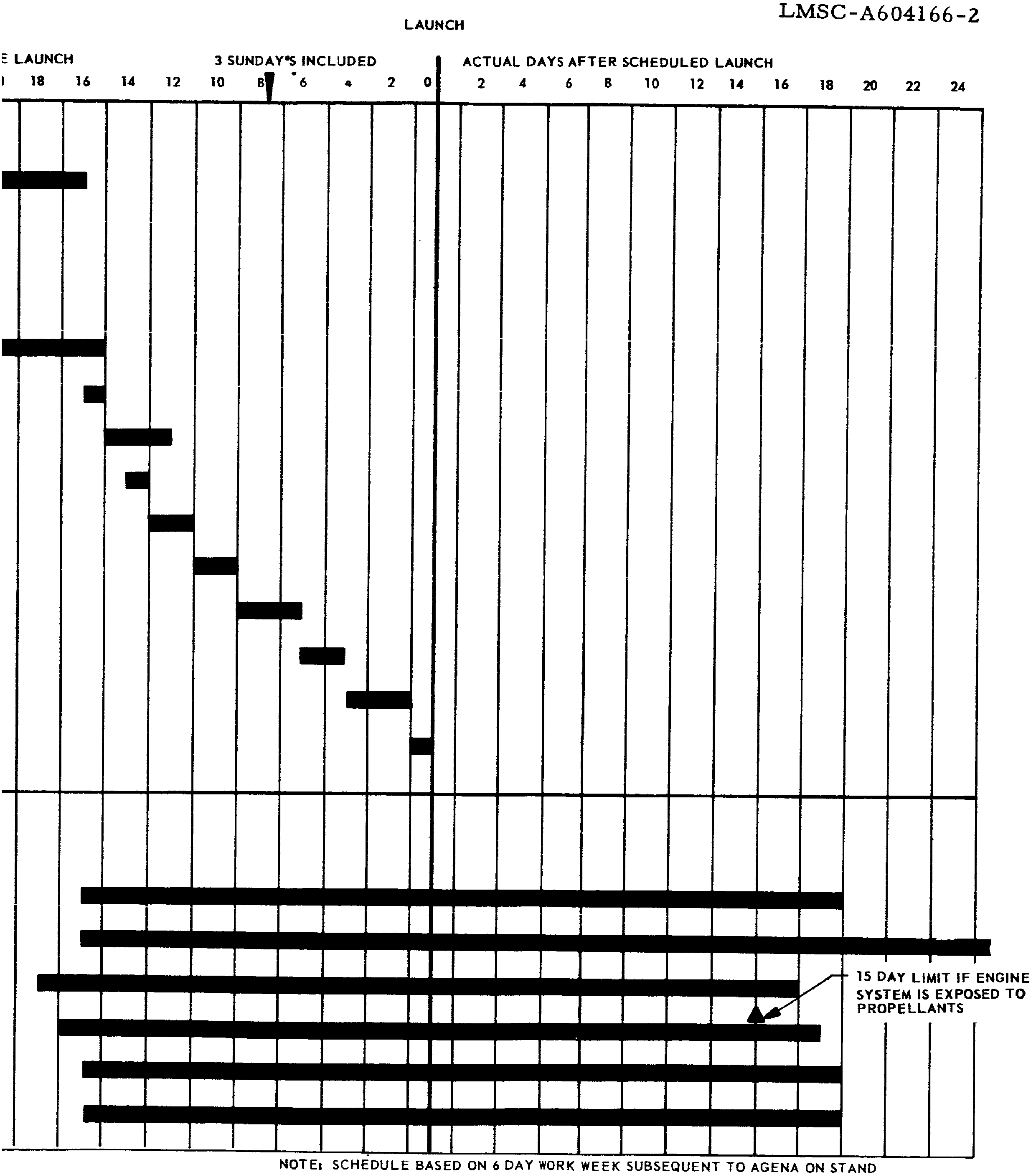


Figure 19-2 Atlas/Agena/Spacecraft Integrated Vehicle Milestones and Agena On-Stand Hold Capability

19.2.2 TAT/Agena (WTR)

The prelaunch test sequence for the TAT/Agena D/spacecraft configuration is shown in Fig. 19-3. Time spans for these tests are defined in Fig. 19-4.

Assembly building testing for the TAT, Agena D, and spacecraft vehicles consists primarily of those tests, validations, and alignments necessary to insure vehicle readiness following shipment to the launch base and for subsequent launch pad activities. The spacecraft/Agena D interstage structure is checked for fit and alignment at this time. For those programs requiring the use of a prototype spacecraft or simulator during launch pad testing, this item is coordinated with the flight article for the purpose of simulated launch data evaluation.

Launch pad testing is performed initially on the separate TAT, Agena D, and spacecraft vehicles. These tests include revalidation of critical vehicle subsystems and checkout of launch pad and blockhouse AGE. Spacecraft/Agena D AGE compatibility checks are performed to provide for the verification of spacecraft measurements transmitted through the Agena umbilical during subsequent combined vehicle testing.

TAT/Agena D/spacecraft combined vehicle tests are performed in two modes, horizontal and vertical.

In the horizontal test mode, the individual vehicles are electrically mated with interconnecting cables but are not physically attached. During this test phase, a flight systems test is conducted which simulates flight events using vehicle power and is performed on an event completion basis. An evaluation of the test data is conducted immediately after the test before proceeding on to the next operational phase.

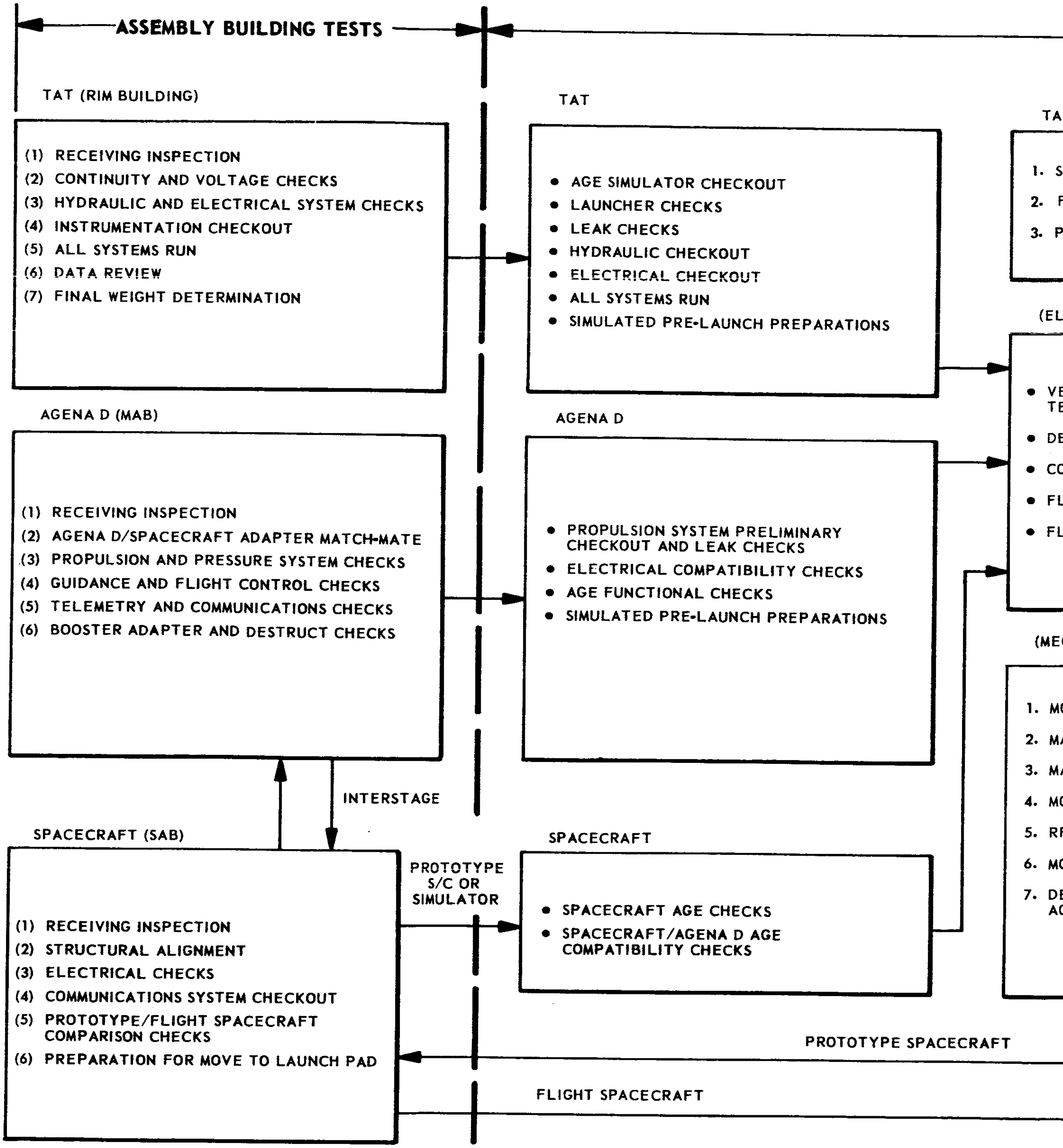
In the vertical test mode the launch vehicle is both electrically and mechanically mated. Two major tests conducted at this time are the mock countdown and the combined RF interference test. The mock countdown, which is a full dress rehearsal of the launch countdown, is performed to verify the ability of the AGE and launch personnel to support the launch countdown, and to insure combined vehicle readiness prior to final R-day prelaunch preparations.

The combined RF interference test consists of RF compatibility tests of the integrated vehicle and range readout tests of the vehicle beacon systems. An evaluation of the data acquired during this test phase provides the basis for the decision to proceed with final R-day prelaunch preparations.

Prelaunch preparations or readiness days for the TAT and Agena D vehicles consist primarily of final flight hardware installations and preflight operations required to verify the readiness of the vehicles for the launch countdown. As indicated in Figs. 19-3 and 19-4, all preparations are completed by the day prior to launch with the exception of propellant tanking, propellant system pressurization, and destruct system arming, all of which are accomplished during the range countdown.

19.3 RANGE COUNTDOWN

The range countdown refers to a series of coordinated operations arranged on an elapsed time basis which are formalized into a procedural document called the "countdown manual", which is prepared jointly by representatives of the spacecraft, launch vehicle, and range agencies, and which is approved by the Launch Operations Working Group (LOWG). Spacecraft inputs to the countdown manual are developed by the responsible Spacecraft Center and the spacecraft launch base organization. Figures 19-5 and 19-6 are typical countdown sequences for Atlas/Agena D/spacecraft (ETR) and TAT/Agena D/ spacecraft configurations (WTR) respectively. Procedures covering abort situations for the vehicle both before and after propellant tanking are also provided as a function of the countdown manual.


ASSEMBLY BUILDING TESTS
TAT (RIM BUILDING)
(1) RECEIVING INSPECTION
(2) CONTINUITY AND VOLTAGE CHECKS
(3) HYDRAULIC AND ELECTRICAL SYSTEM CHECKS
(4) INSTRUMENTATION CHECKOUT
(5) ALL SYSTEMS RUN
(6) DATA REVIEW
(7) FINAL WEIGHT DETERMINATION
TAT
• AGE SIMULATOR CHECKOUT
• LAUNCHER CHECKS
• LEAK CHECKS
• HYDRAULIC CHECKOUT
• ELECTRICAL CHECKOUT
• ALL SYSTEMS RUN
• SIMULATED PRE-LAUNCH PREPARATIONS
AGENA D (MAB)
(1) RECEIVING INSPECTION
(2) AGENA D/SPACECRAFT ADAPTER MATCH-MATE
(3) PROPULSION AND PRESSURE SYSTEM CHECKS
(4) GUIDANCE AND FLIGHT CONTROL CHECKS
(5) TELEMETRY AND COMMUNICATIONS CHECKS
(6) BOOSTER ADAPTER AND DESTRUCT CHECKS
AGENA D
• PROPULSION SYSTEM PRELIMINARY CHECKOUT AND LEAK CHECKS
• ELECTRICAL COMPATIBILITY CHECKS
• AGE FUNCTIONAL CHECKS
• SIMULATED PRE-LAUNCH PREPARATIONS
INTERSTAGE
SPACECRAFT (SAB)
(1) RECEIVING INSPECTION
(2) STRUCTURAL ALIGNMENT
(3) ELECTRICAL CHECKS
(4) COMMUNICATIONS SYSTEM CHECKOUT
(5) PROTOTYPE/FLIGHT SPACECRAFT COMPARISON CHECKS
(6) PREPARATION FOR MOVE TO LAUNCH PAD
PROTOTYPE S/C OR SIMULATOR
SPACECRAFT
• SPACECRAFT AGE CHECKS
• SPACECRAFT/AGENA D AGE COMPATIBILITY CHECKS
PROTOTYPE SPACECRAFT
FLIGHT SPACECRAFT

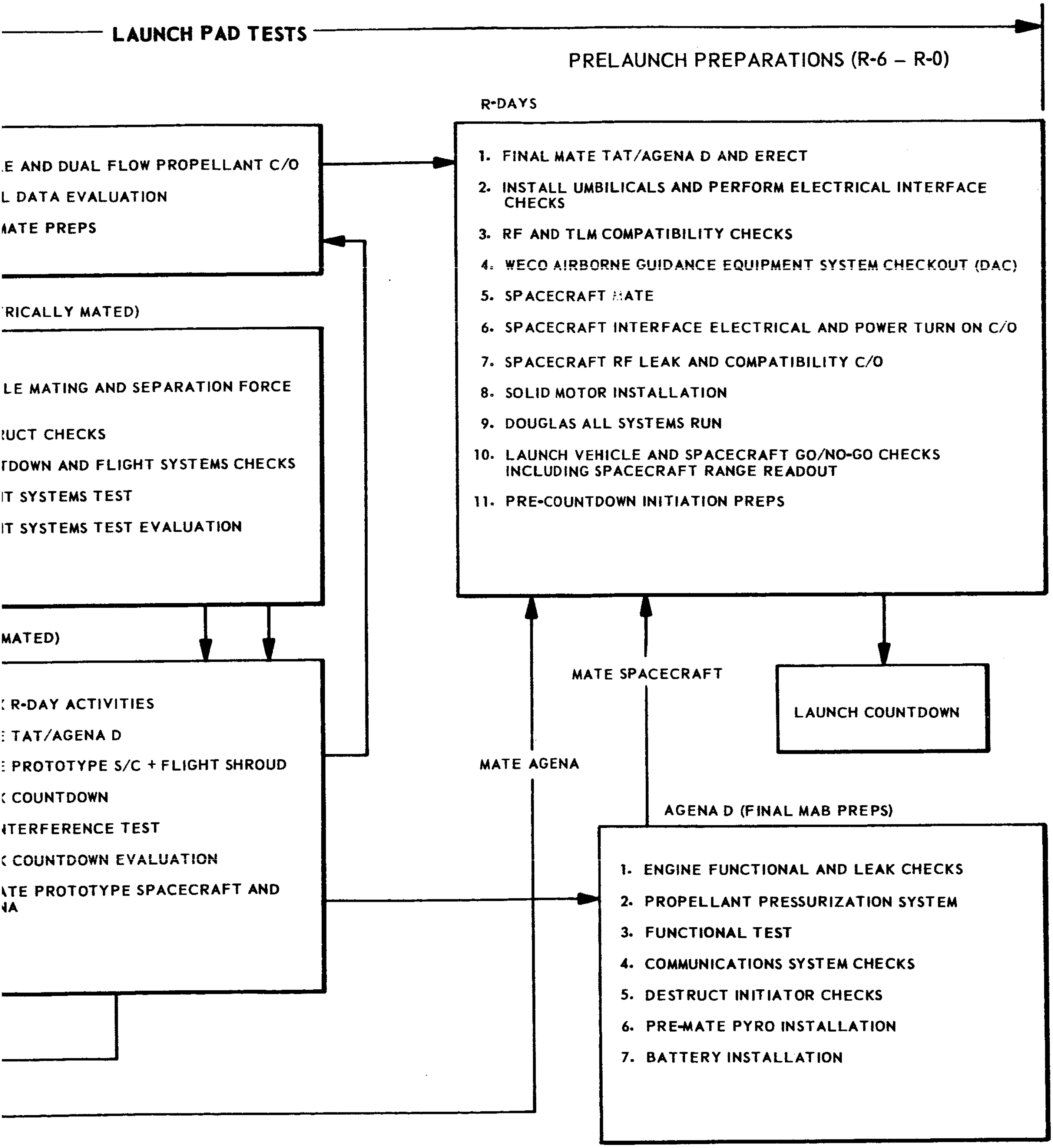


Figure 19-3 Typical TAT/Agena D/Spacecraft Prelaunch Test Sequences

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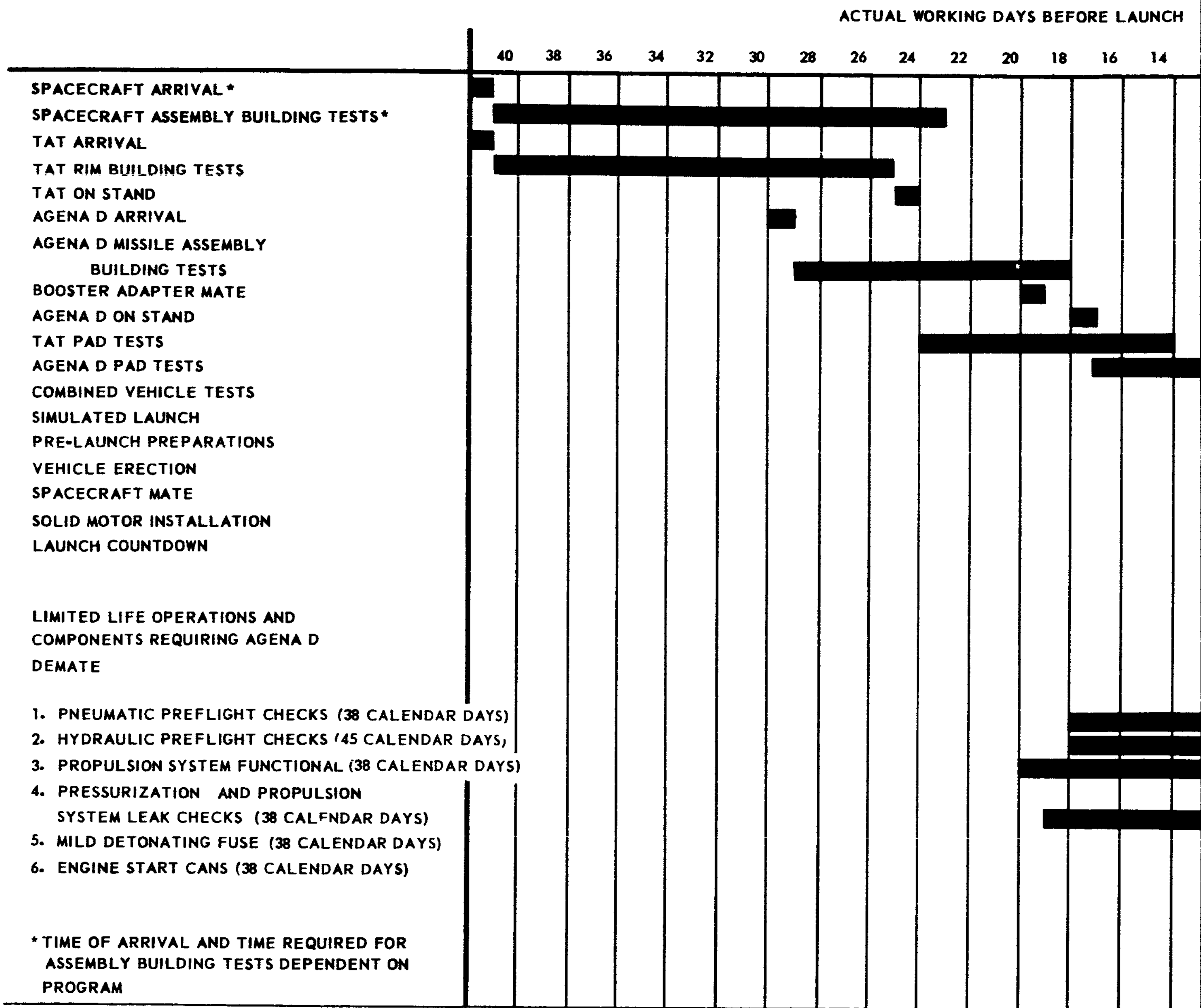
ACTUAL WORKING DAYS BEFORE LAUNCH
40
38
36
34
32
30
28
26
24
22
20
18
16
14
SPACECRAFT ARRIVAL*
SPACECRAFT ASSEMBLY BUILDING TESTS*
TAT ARRIVAL
TAT RIM BUILDING TESTS
TAT ON STAND
AGENA D ARRIVAL
AGENA D MISSILE ASSEMBLY BUILDING TESTS
BOOSTER ADAPTER MATE
AGENA D ON STAND
TAT PAD TESTS
AGENA D PAD TESTS
COMBINED VEHICLE TESTS
SIMULATED LAUNCH
PRE-LAUNCH PREPARATIONS
VEHICLE ERECTION
SPACECRAFT MATE
SOLID MOTOR INSTALLATION
LAUNCH COUNTDOWN
LIMITED LIFE OPERATIONS AND COMPONENTS REQUIRING AGENA D DEMATE
1. PNEUMATIC PREFLIGHT CHECKS (38 CALENDAR DAYS)
2. HYDRAULIC PREFLIGHT CHECKS (45 CALENDAR DAYS)
3. PROPULSION SYSTEM FUNCTIONAL (38 CALENDAR DAYS)
4. PRESSURIZATION AND PROPULSION SYSTEM LEAK CHECKS (38 CALENDAR DAYS)
5. MILD DETONATING FUSE (38 CALENDAR DAYS)
6. ENGINE START CANS (38 CALENDAR DAYS)
*TIME OF ARRIVAL AND TIME REQUIRED FOR ASSEMBLY BUILDING TESTS DEPENDENT ON PROGRAM

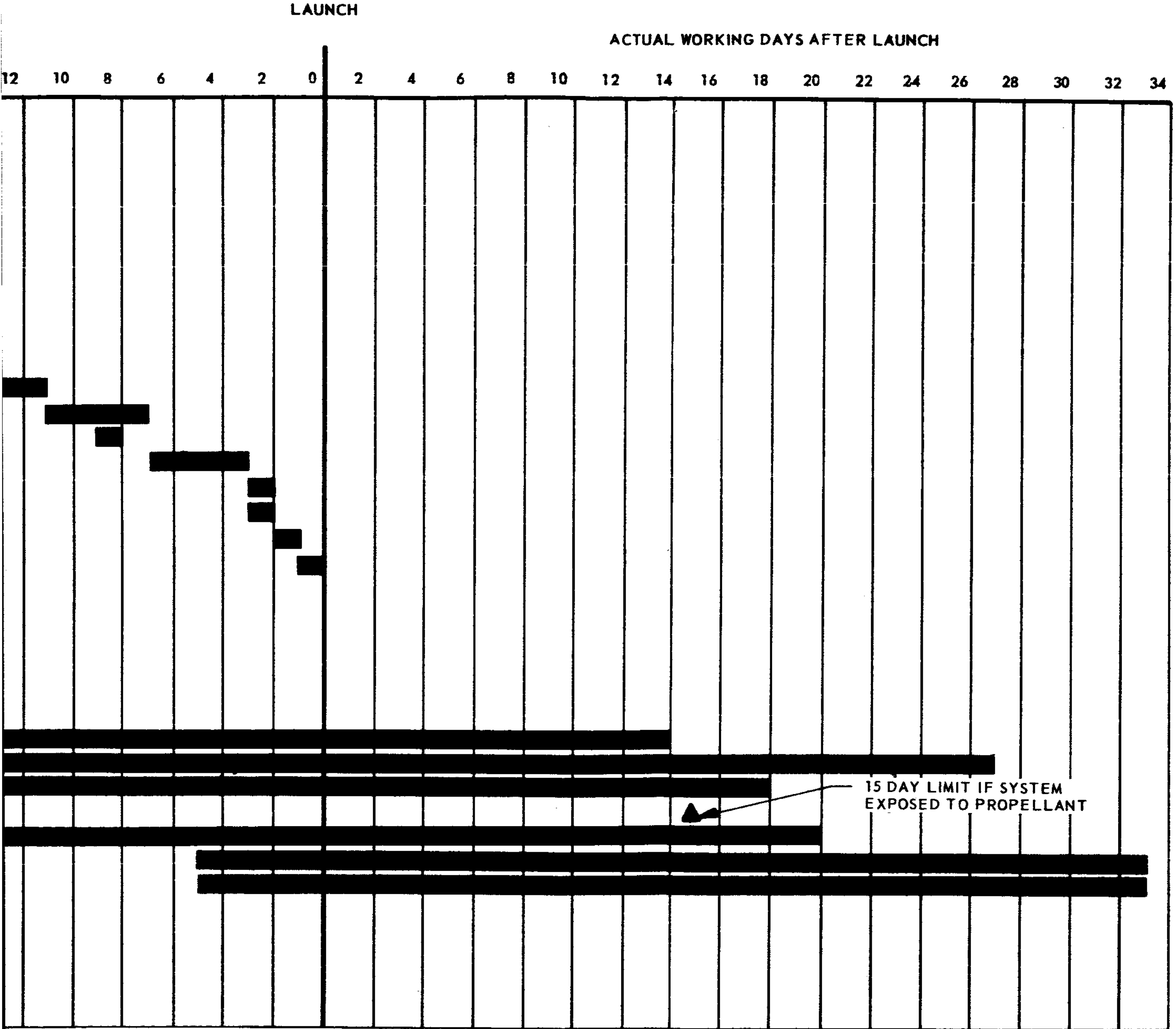

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Figure 19-4 TAT/Agena D/Spacecraft Integrated Vehicle Milestone Chart and Agena On-Stand Hold Capability

	RF SILENCE; START RANGE COUNT; RF SILENCE
TIME-MINUTES	340; 320; 300; 280; 260; 240; 220; 200; 180
TEST CONDITIONS	RED; AMBER; RED; AMBER
PROPULSION	IGNITER INSTALLATION; IGNITER DETONATOR INSTALLATION; CHECK REGS
TELEMETRY	
RANGE SAFETY	MECHANICAL INSTALLATION RED BOX; RSC CHECK; ELECTRICAL CONNECTOR RED BOX
AUTO PILOT	WARMUP; TE
HYDRAULIC	HYDRAULIC CHECK
MOD III GUIDANCE	
LANDLINE INSTRUMENTATION	GROUP I ON SLOW
LO_2 CONTROL	TOPPING GN_2 STORAGE
FUEL CONTROL	F-D; LEAK CHECK
PRESSURIZATION	STAGE I
PNEUMATICS	ENGINE CONTROL SUPPLY
WATER SYSTEMS	18" AND 36" WATER PUMP ON (2 PUMPS ONLY)
MSL ELECTRICAL	
GROUND SUPPORT	CRITICAL POWER AND POD COOLING ON
GROUND SUPPORT	

GD/A LV-3A

AIRFRAME	START CLOSEOUT; FINAL CLOSEOUT
UDMH TANK	PREP
IRFNA TANK	PREP
FU_2 PRESSURIZATION	PREP
HELIUM PRESSURIZATION	PREP
ELECTRICAL	PREP; POWER CHECK; REMOVE TEST PLUG
PYROTECHNIC	DESTRUCT PYRO TEST; RSC TEST; NO VOLT HOOKUP
GUIDANCE	CONSOLE CHECKS; POWER ON; HORIZON SENSOR GYRO FLIGHT CONTROL CHECKS; VELOCITY METER CHECKS
TELEMETER	G.S. AND CONSOLE CHECKS; TELEMETRY CHECKS
BEACON	G.S. AND CONSOLE CHECKS; BEACON CHECKS
AGE	POWER PREP CHECKS
AGENA A-C POWER	ON
	340; 320; 300; 280; 260; 240; 220; 200; 180

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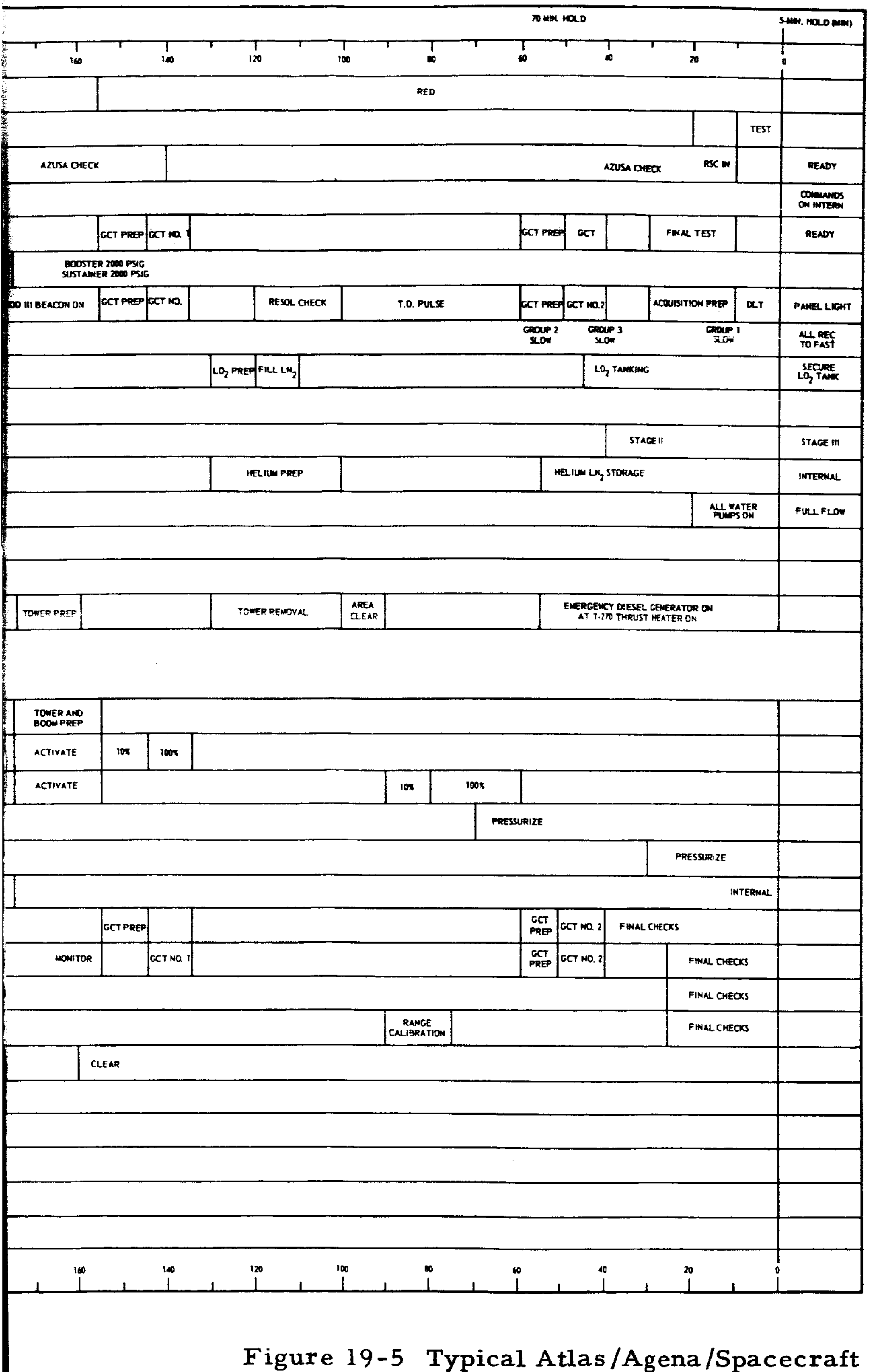

Figure 19-5 Typical Atlas/Agena/Spacecraft Launch Countdown (ETR)

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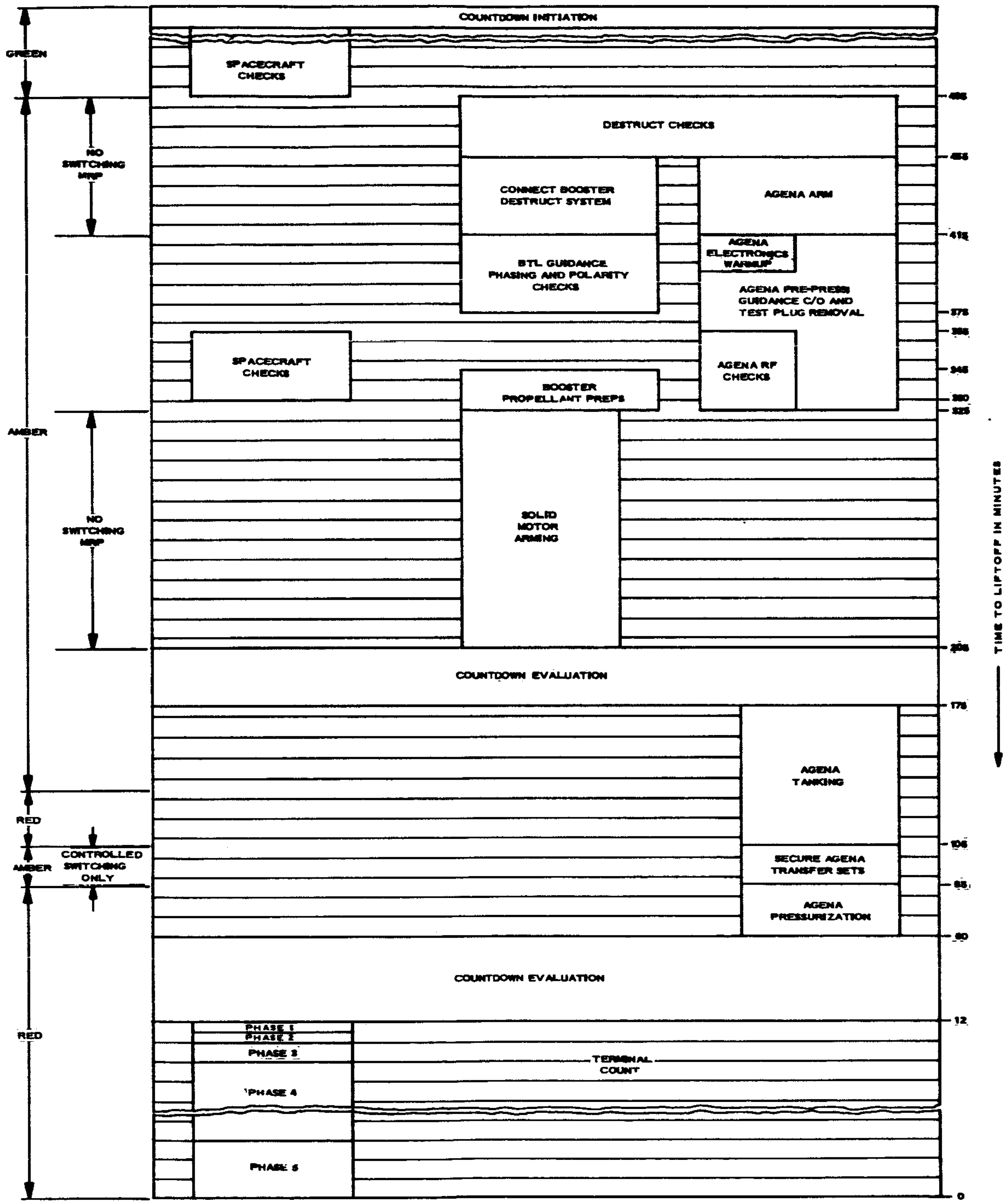

Figure 19-6 Typical TAT/Agena D/Spacecraft Launch Countdown (WTR)

19.4 EQUIPMENT VALIDATION TIME RESTRAINTS

A launch may be delayed or aborted because equipment operating life, equipment "maximum hold time," or subsystem validation time period is exceeded. Limited Operating Life (LOL) equipment of the Agena is identified in LMSC Specification 1414870 and includes those items which are subject to wear out by usage, or on which operating data are required for evaluation. If the ground life limit is exceeded, program engineering will recommend equipment disposition. Equipment such as batteries and pyrotechnics have "maximum hold time" and if the hold time is exceeded the parts are replaced. (Recharge secondary type batteries)

Between two and three weeks prior to launch, the Agena subsystems are subjected to tests consisting primarily of leak checks and functional checks. Figures 19-2 and 19-4 show the hold time capability of Atlas/Agena and TAT/Agena, respectively. If the validation time periods are exceeded the booster/Agena must be demated. Certain guidance system components, such as the velocity meter and the inertial reference package, have a gradual scale factor change in time and thus require recalibration at specified intervals, i.e., 45 days in the case of the velocity meter.

19.5 ENVIRONMENTAL LAUNCH RESTRAINTS

Two variable atmospheric conditions that may cause launch delay are abnormal winds aloft and excessive cloud cover. The determination of winds aloft and cloud cover conditions, evaluation of effects and procedures for go/no-go recommendations are discussed in the following paragraphs.

19.5.1 Winds Aloft Restraints

Launches may be subject to delay due to abnormal distribution of wind velocities and shears in the region above and downrange from the launch pad along the vehicle's ascent trajectory, and ground wind gusts at the launch complex. Some wind patterns can put aerodynamic loads on the vehicle severe enough to cause breakup, or flight path and attitude deviations beyond the control system recovery capabilities. To ensure that the vehicle will not encounter a wind environment that exceeds the design limits, winds aloft and ground wind condition are monitored prior to launch, and the vehicle's response to wind conditions as they exist at the launch site is determined. Based on the winds analysis, a "go" or "no-go" recommendation for launch is made according to the procedure outlined in Fig. 19-7, which presents a block diagram of the flow of wind data from the launch site to the Contractor.

19.5.2 Cloud Cover Restraints

A second condition which may delay launch is excess cloud cover in the areas scanned by the Agena horizon sensors during the first burn phase of the flight. Under certain conditions, clouds in this area may change the infrared gradient characteristic "seen" by the sensors and cause the guidance system to sense a false Agena horizon, and could lead to unacceptable injection dispersions. The cloud cover is monitored (by cloud "mapping") in these critical regions, and probable effects on injection accuracy are determined in computer simulation studies just prior to launch. Should the predicted effect be too great, the launch may be delayed. (Agena guidance sensitivities to cloud cover during second burn are not of sufficient magnitude to require cloud cover analysis for later portions of the flight.) Figure 19-8 presents a description of the data and the communications flow of the computed cold cloud parameters, including a map of the area to be examined on a typical ETR launch.

19.6 RANGE SUPPORT

19.6.1 Eastern Test Range

Figure 19-9 presents the ETR tracking and telemetry coverage for typical Atlas/Agena probe launches. Range safety requirements generally limit the launch azimuth corridor within which the particular space vehicle must fly.

Command destruct capability is normally required in the event the vehicle does not remain in the corridor. Tracking and telemetry coverage is shown in Figure 19-9 for the 100 nm parking orbits for typical space probe missions. Cape Kennedy, Grand Bahama Island, Grand Turk and Antigua tracking and telemetry stations provide coverage of Atlas/Agena ascent through Agena first burn. Location of second burn is variable dependent upon the missions targeting requirements. In the illustrated case the Agena second burn and spacecraft separation events are covered by the tracking and telemetry stations located at Ascension Island and Pretoria, South Africa. Also shown is a telemetry ship typical of tracking and telemetry ships utilized to supplement land station capabilities.

For an earth orbiting spacecraft mission utilizing a single Agena burn, the Cape Kennedy, Grand Bahama Island, Grand Turk and Antigua tracking and telemetry stations usually provide data coverage for the Atlas and Agena boost portions of the flight. In the case of a dual burn mission, the above ETR stations provide coverage through Agena first burn and the Pretoria or other suitable stations may provide coverage of the Agena second burn sequence and spacecraft separation. Figure 19-10 and 19-11 present typical range coverage for a satellite mission.

The up-range tracking stations relay data to a Cape Kennedy computer in real time for calculation of downrange acquisition parameters, predicted impact points, and quick-look vehicle performance analysis. Significant vehicle telemetered events are monitored in real time and the time of occurrence is relayed to Cape Kennedy for quick look vehicle performance analysis. The Range Safety Officer utilizes impact predictions and other data in maintaining surveillance of the flight. In the event the vehicle becomes a potential hazard, the Range Safety Officer normally has the capability of commanding vehicle destruct up to the time that the Agena attains orbital conditions.

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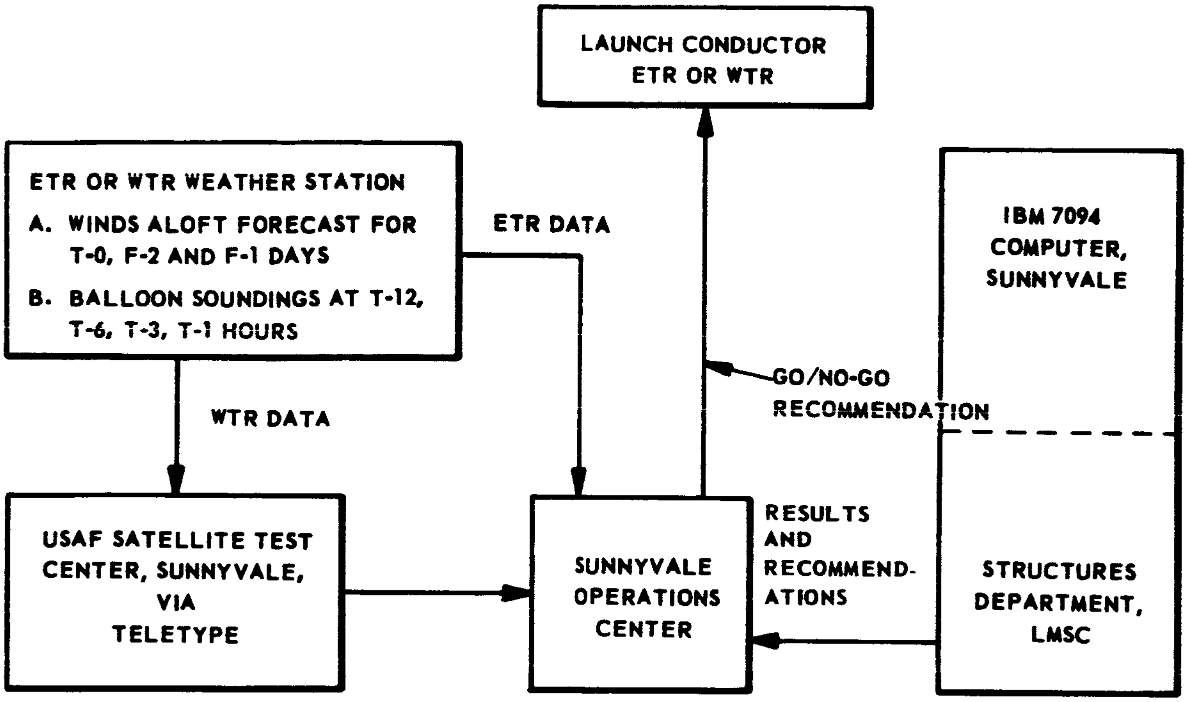
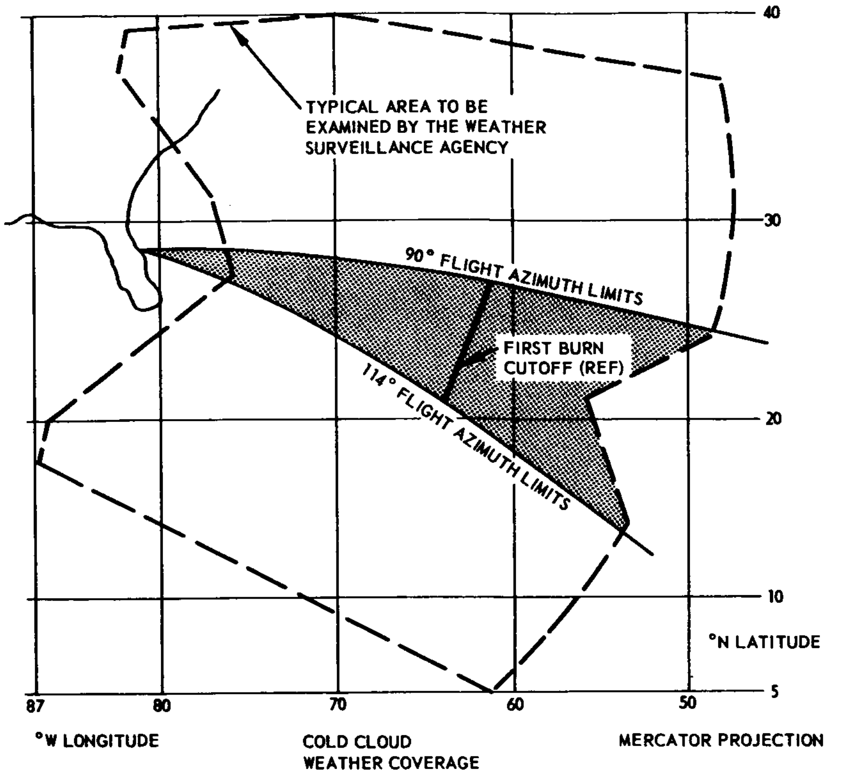


WINDS ALOFT "GO/NO-GO" RECOMMENDATION PROCEDURE *		
BENDING MOMENT (PERCENT)	CONTROL CAPABILITY (PERCENT)	ACTION
90 OR GREATER	90 OR GREATER	RECOMMEND "NO-GO"
80 – 90	80 – 90	RECOMMEND "GO" BASED ON F-1, T-14, T-8, T-5, T-1 SOUNDINGS. ADDITIONAL SOUNDING WILL BE MADE AT T-3. RECOMMEND "GO" BASED ON T-5 AND T-3 SOUNDINGS PROVIDED STRUCTURAL RESPONSE IS DECREASING.
70 – 80	70 – 80	RECOMMEND "GO" BASED ON F-1, T-14, T-8, T-5, T-1 SOUNDINGS. ADDITIONAL SOUNDING WILL BE MADE AT T-3. RECOMMEND "GO" BASED ON T-5 AND T-3 SOUNDINGS PROVIDED STRUCTURAL RESPONSE IS UNCHANGING OR DECREASING.
70 OR LESS	70 OR LESS	RECOMMEND "GO", REQUEST T-3 SOUNDING

* LMSC-A306008

Figure 19-7 Winds Aloft "Go/No-Go" Recommendation Procedure



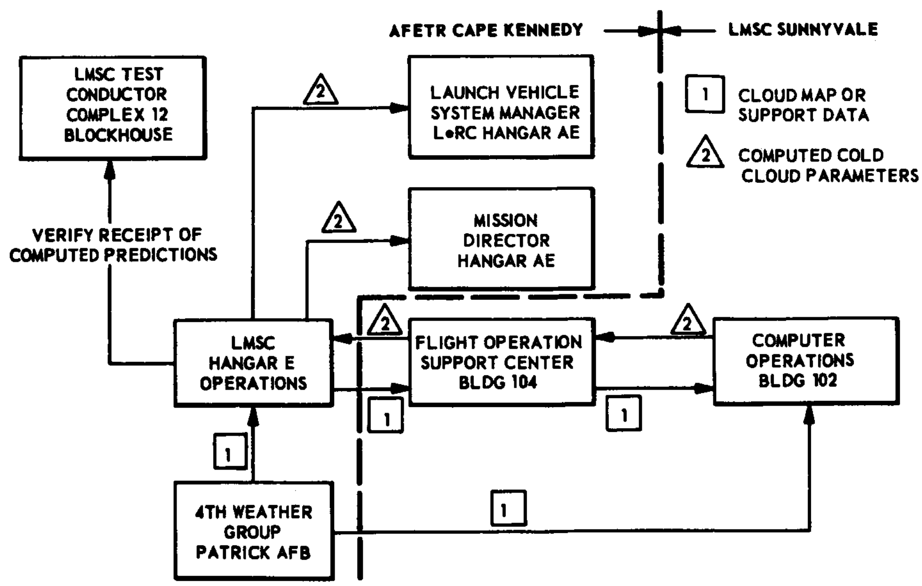


Figure 19-8 Cold Cloud Data and Communications Flow

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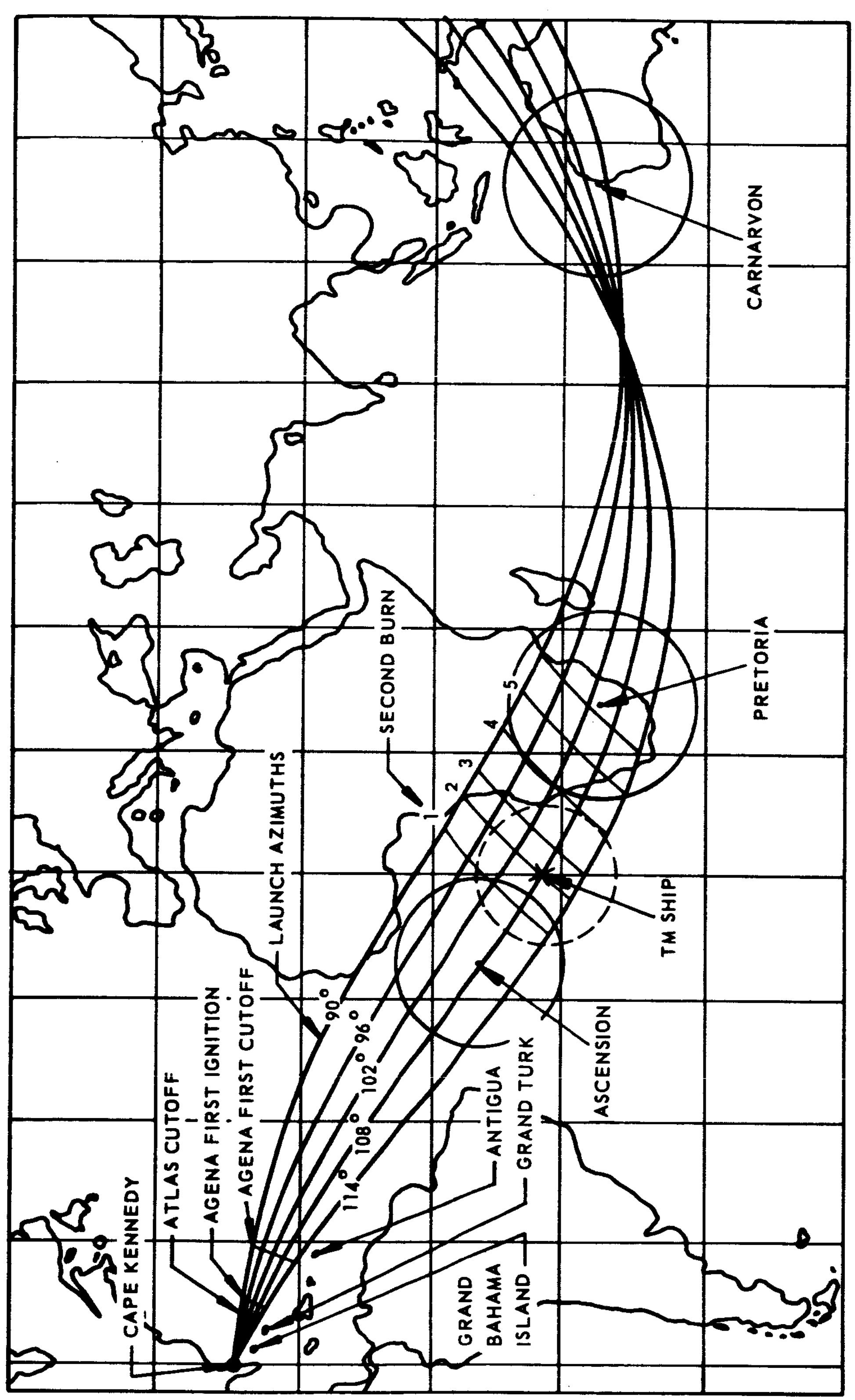

Figure 19-9 Atlas/Agena/Spacecraft Ascent Tracking and Telemetry Coverage

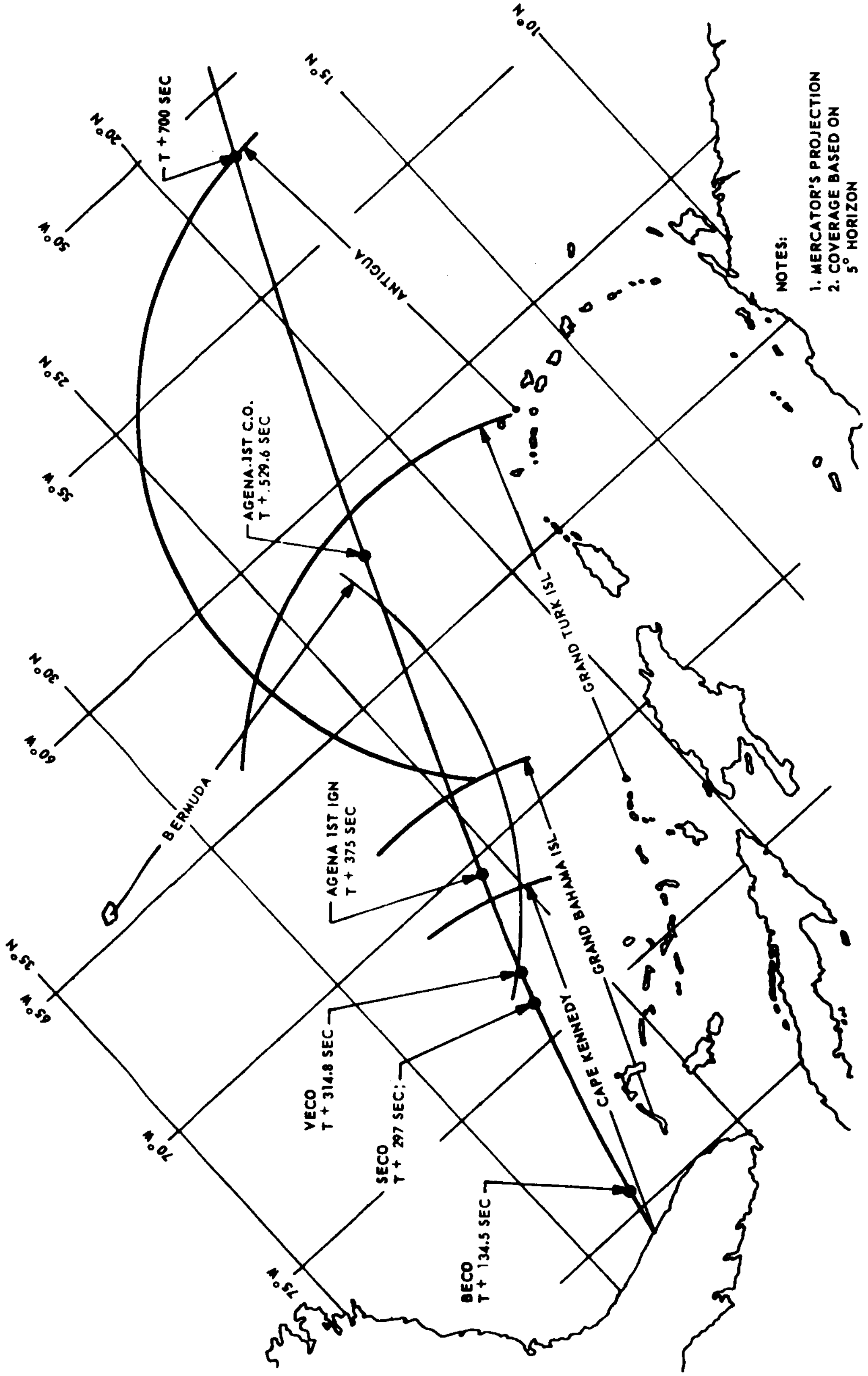

Figure 19-10 Typical ETR Tracking Coverage From Liftoff Through Agena First Burn (EGO Mission)

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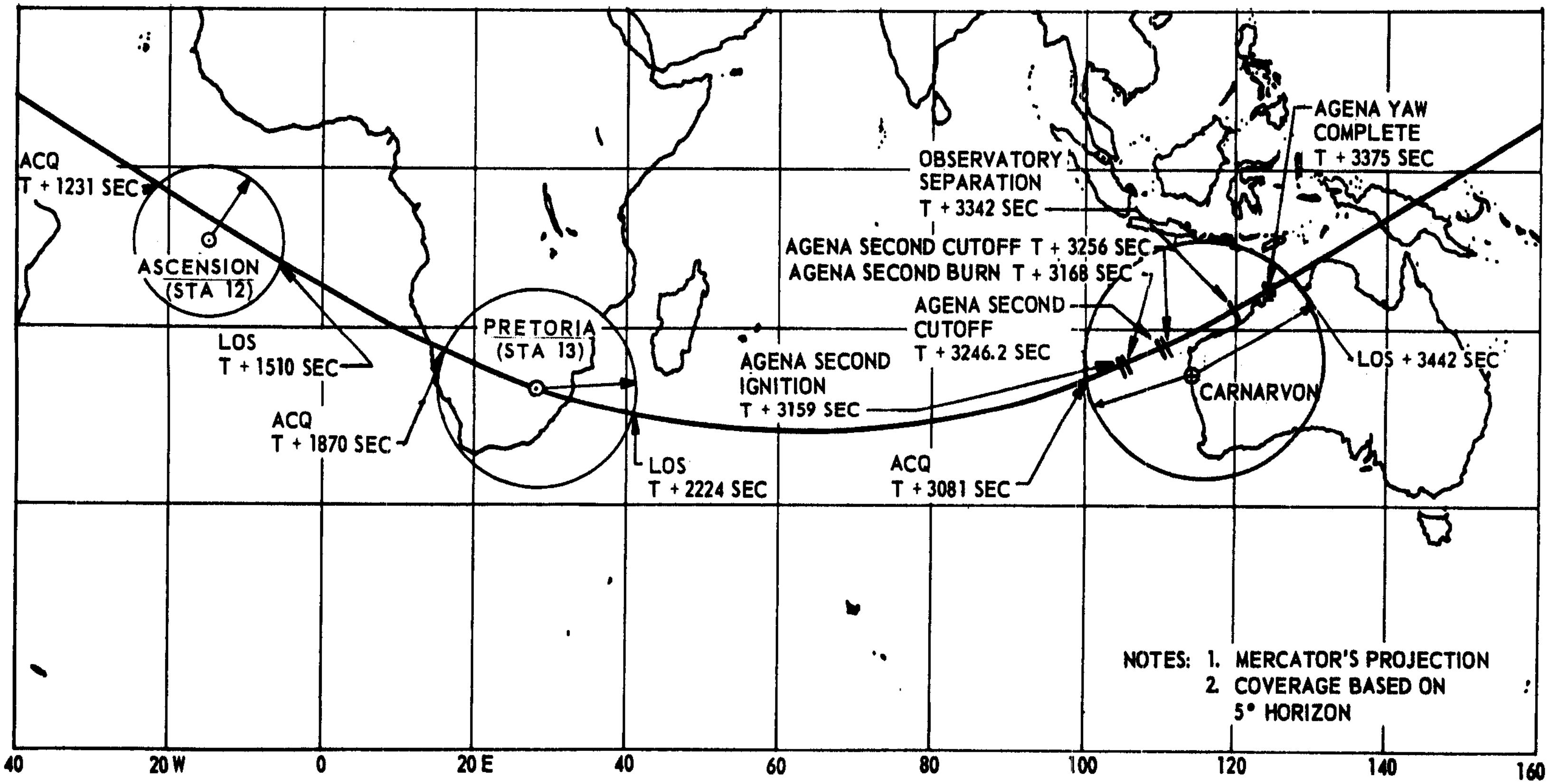

Figure 19-11 Typical Radar Tracking Coverage During Transfer Orbit, Agena Second Burn, and Agena/Spacecraft Separation (EGO Mission)

The real time data are transmitted from stations as far as Antigua via submarine cable. Ascension, Pretoria, and ship "near" real time data are transmitted via single side band radio. The original tracking and telemetry magnetic tapes from the downrange stations are returned to Cape Kennedy via aircraft for processing. Cape Kennedy Space Center reduces the data and then delivers copies to the range users for their use in performance analysis, etc. Figure 19-12 and 19-13 depicts the tracking data flow from the ETR and the Carnarvon Station.

19.6.2 Western Test Range

Range support for TAT/Agena D/Spacecraft orbital missions launched from the WTR will be provided by land-based stations supplemented by range ships as required. Vehicle launch azimuths between 175 degrees and 220 degrees are considered to be acceptable for range safety purposes insofar as minimizing likelihood of vehicle impact on populated land masses. Launch azimuths outside of this range will require obtaining a waiver from the WTR Range Safety division. (Waiver applications usually are supported by evaluations of impact and kill probabilities.)

For a typical TAT/Agena D/spacecraft mission (Fig. 19-14), the vehicle is tracked by optical systems at Vandenberg Air Force Base, Point Arguello, Point Mugu, and San Nicholas Island. Land based radar capability exists at Pt. Arguello, Point Mugu, and San Nicolas Island. Tracking and/or telemetry ships are positioned downrange as required. For mission utilizing an Agena D single burn, the data acquired by these stations will be transmitted to the responsible NASA Control Center for computation of acquisition data for the orbital tracking and data acquisition network. For dual burn missions, acquisition data computed at the WTR will be transmitted to ETR Station 13 (Fig. 19-15) which normally provides radar tracking and telemetry coverage for Agena D second burn and spacecraft injection events. The injection parameters are then transmitted to the NASA Control Center for use by the orbital network as stated above.

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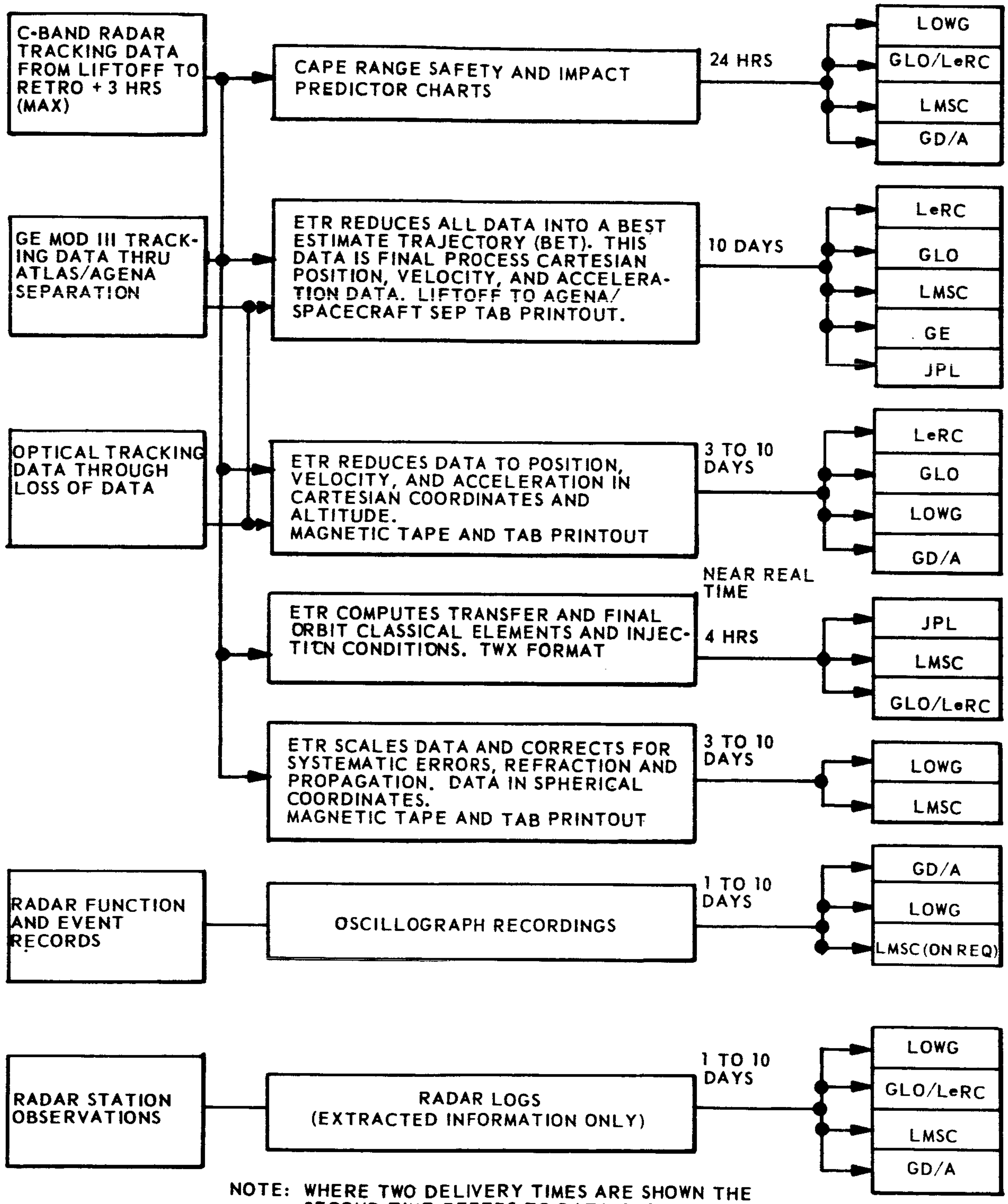

NOTE: WHERE TWO DELIVERY TIMES ARE SHOWN THE SECOND TIME REFERS TO DATA FROM DOWN-RANGE STATIONS.

Figure 19-12 ETR Tracking Data Flow

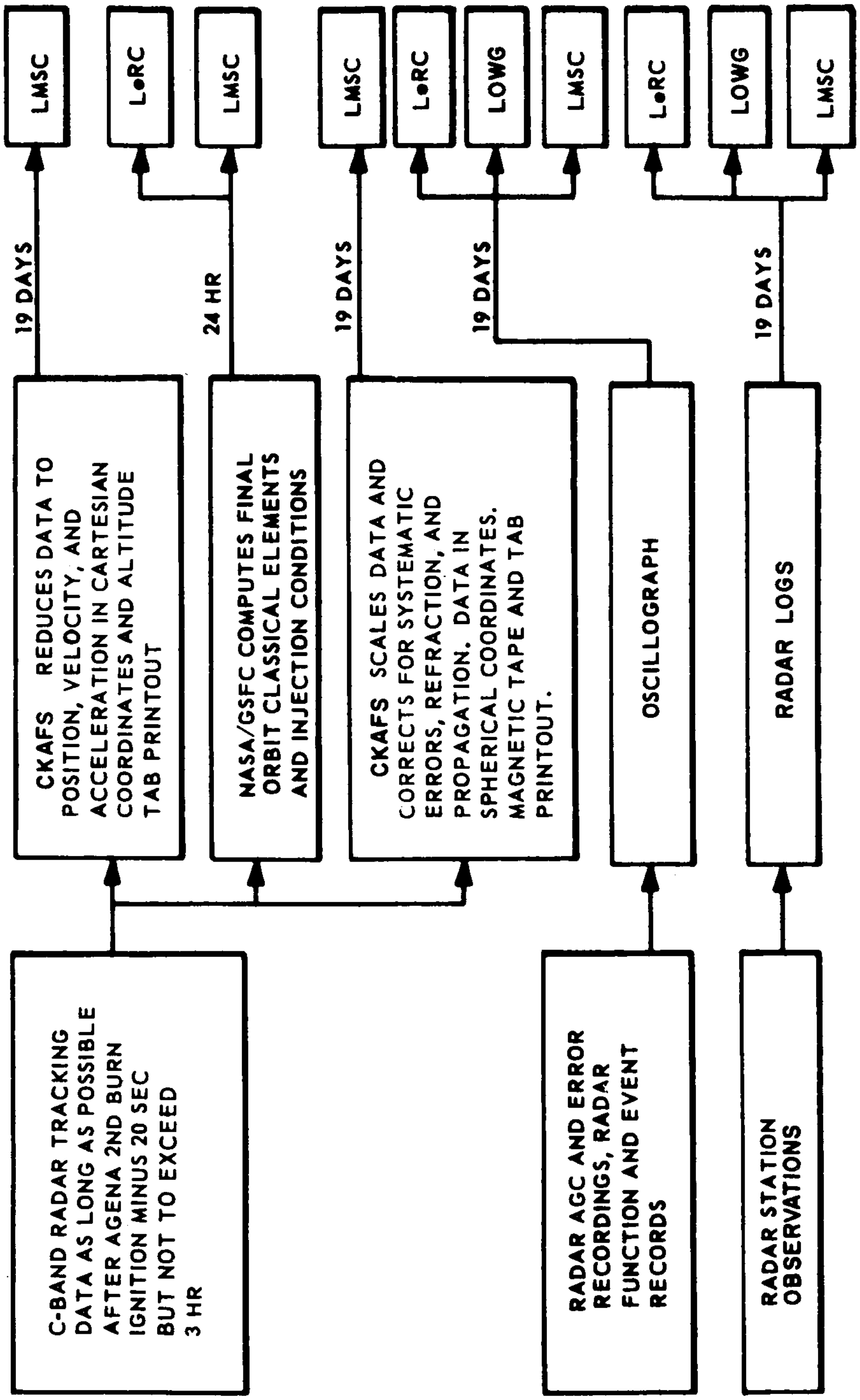

Figure 19-13 Carnarvon Tracking Station Data Flow

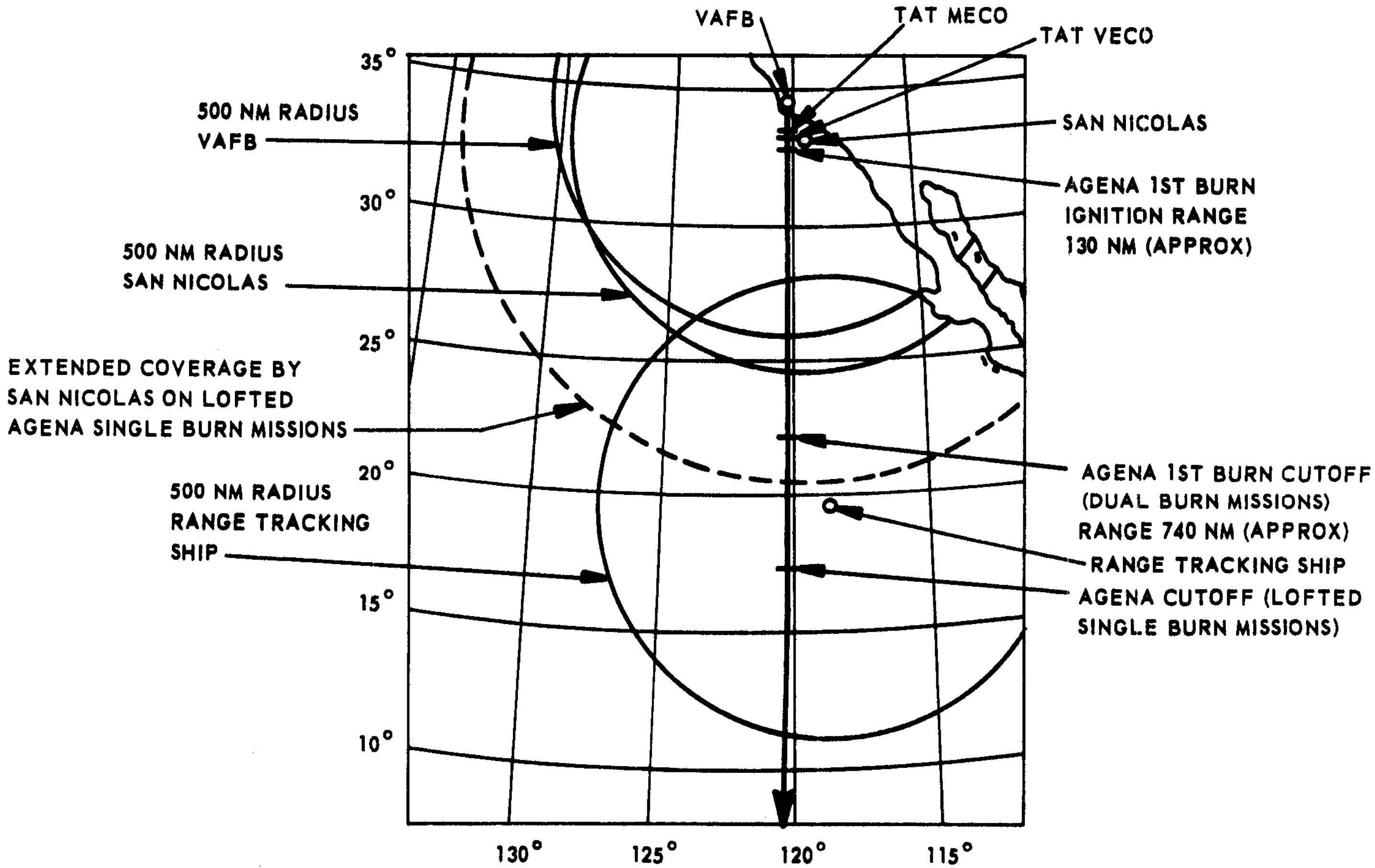

Figure 19-14 TAT/Agena D/Spacecraft Typical WTR Ascent Tracking Coverage

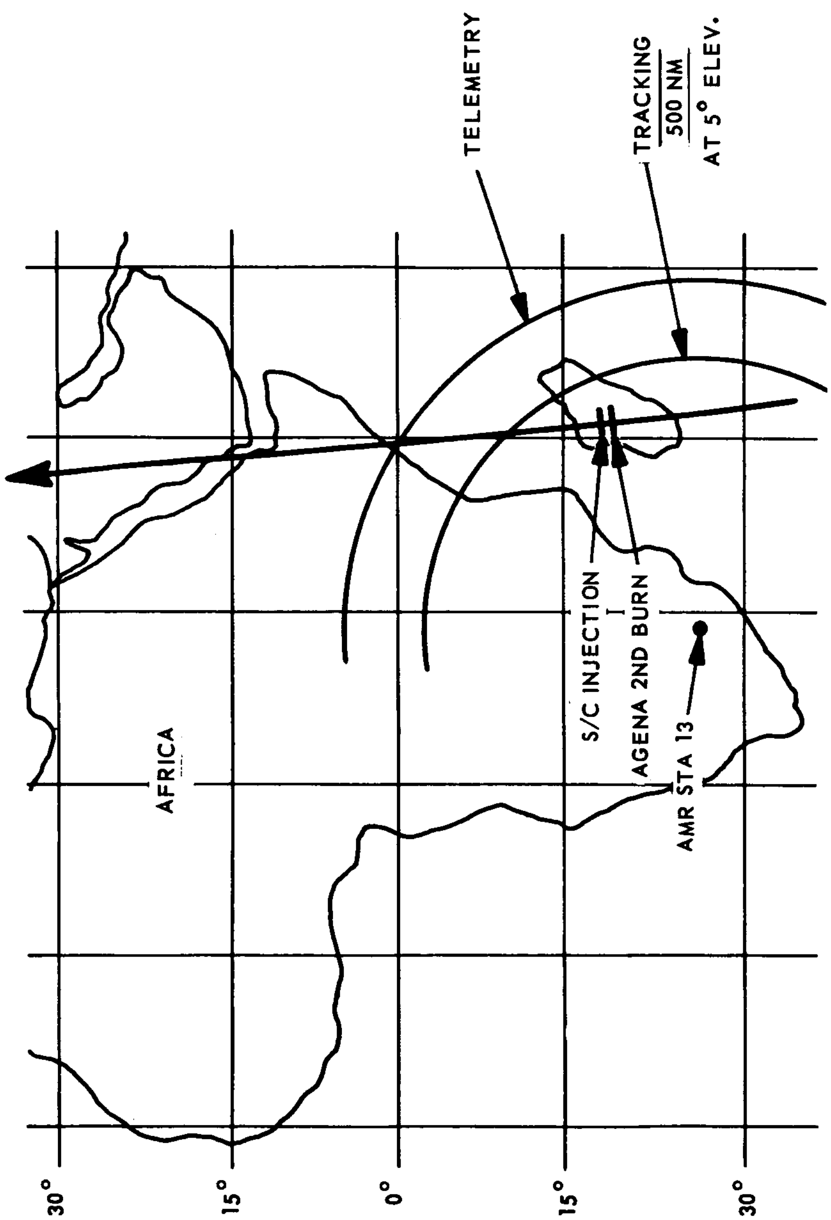

Figure 19-15 Tracking Coverage for Agena D 2nd Burn and Spacecraft Injection (WTR Launch)

Telemetry data are received, and recorded, at Point Arguello, Point Mugu, San Nicolas Island, ETR Station 13, and on the range tracking ships. These data are utilized for post flight analysis of vehicle performance.

Range communications at the WTR provide for (1) the coordination of data acquisition and transmittal of data within the range area, and (2) the integration of range areas into a coordinated range complex. Landline and teletype circuits are available between Vandenberg Air Force Base, Point Arguello, Point Mugu, and San Nicolas Island. RF links are utilized for communications between the range tracking ships and the land based stations. Communications between the WTR and ETR Station 13 are handled by landline to Cape Kennedy and by RF links from Cape Kennedy to Station 13.

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SECTION 20
TYPICAL AGENA MISSION PROGRAMMING

20.1 GENERAL

The other sections of this catalog have described the Agena capability, restraints, available hardware, procedures, and techniques which may be applied to meet mission requirements. This section presents a discussion of the overall programming aspects. The objective will be to describe on a project basis, the matrix of activity out of which a qualified NASA Agena mission vehicle will be delivered to the launch pad ready for mating with the booster vehicle and the particular spacecraft. Requirements affecting the spacecraft program are described separately (par. 20.4, 20.5) after discussions of the Agena program establish a suitable frame of reference. The principal coordinating documentation generated for the typical mission is also discussed.

To the reader unfamiliar with the procurement pattern established for Agena D, it will be helpful to note that the basic or standard vehicle is produced by LMSC under a specific Air Force contract. Using agencies, Air Force or NASA, purchase these basic vehicles and then make separate mission procurements for applications. This catalog is concerned primarily with the LMSC programming of the NASA satellite and probe mission procurement, where the Agena D basic vehicle is furnished as government equipment for augmentation or adaptation to perform the particular mission. Figure 20-1 summarizes the prevailing arrangement for NASA satellite and probe missions.

The programming of each NASA Agena mission involves both hardware and software activity by LMSC. The integration of the hardware activities with mission and supporting analyses (software) is summarized in Fig. 20-2. Characteristically, overall mission requirements and technical direction originate with NASA. Early in the program cycle, the mission spacecraft requirements and restraints on the launch vehicle system are to be formalized in a "restraints document"* originated by the Spacecraft Center. These restraints will define the mission, together with a definition of spacecraft interface characteristics, operational restrictions, and operational tolerances. As this document serves as a basis to LeRC for initiating action to procure the mission launch vehicle adaptation and integration contract, it should be received by LeRC no less than 22 months before desired launch.** Consultations with LeRC concerning the preparation of the "Restraints Document" are strongly recommended. LMSC-programmed activity will be structures to implement the mission requirements as defined by these spacecraft restraints.

A schedule summary for the typical NASA Agena mission program is shown in Fig. 20-3, which illustrates a nominal 24 months from mission project approval to launch and a nominal 12 months from contract go-ahead to launch. (This schedule is predicated on the basis that no major peculiar equipment development is required; i.e., shrouds, adapters for spacecraft or vehicles, Agena forward equipment racks, Agena horizon sensor systems, etc.) Subsequent paragraphs in this section will delineate in greater detail the vehicle hardware activities at each stage in the program.

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*"Agena Mission Standard Requirements and Restraints Document, Vol. 1 - Format and Vol. 11 - Instructions" LMSC SP3805-64-1. Also, see para. 20.4.1, page 20-6, herein.

**The required date will vary with mission complexity; i.e., the amount and kind of peculiar hardware, special analytical studies and length of definition phase required.

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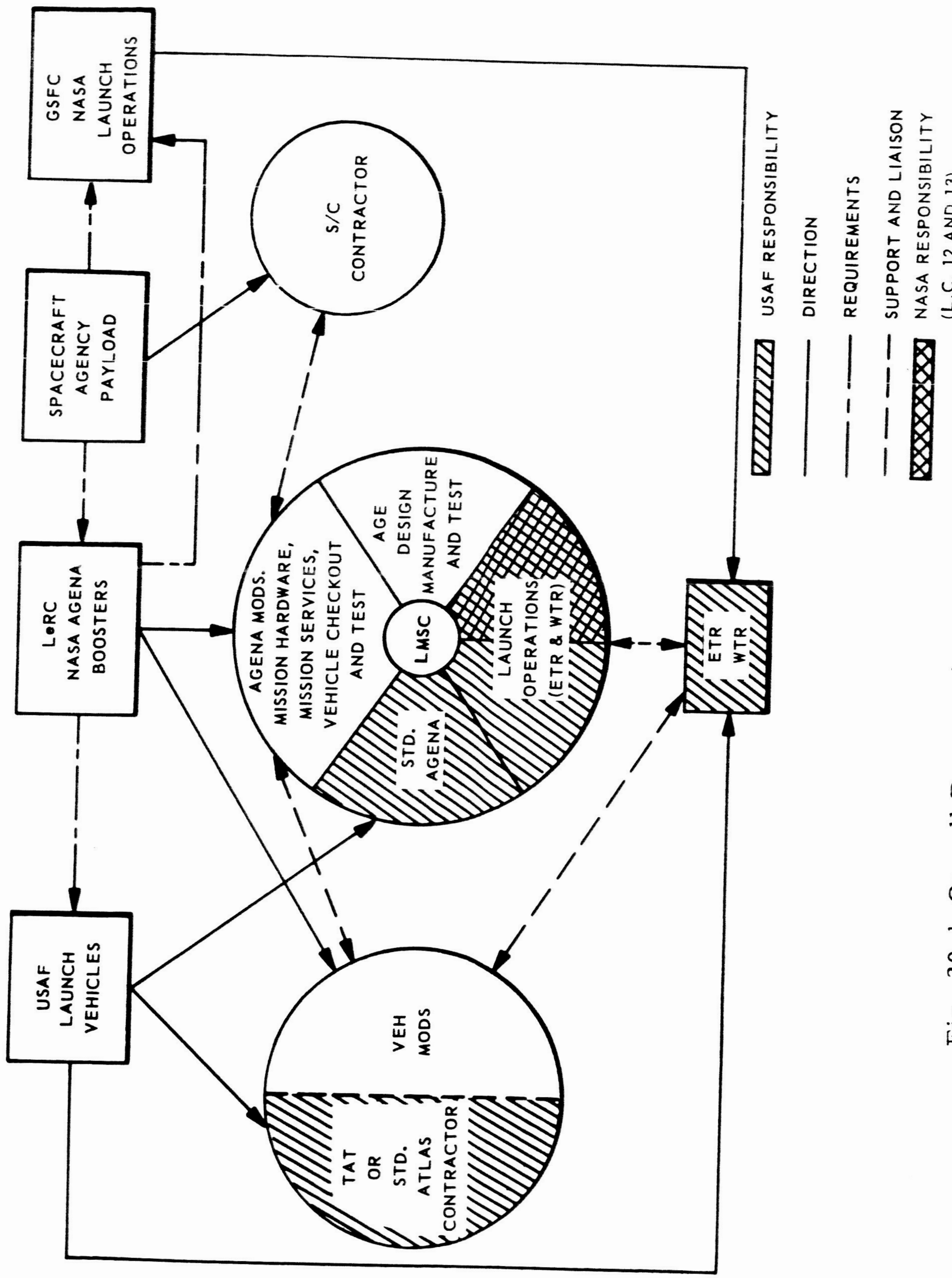

Fig. 20-1 Overall Program Arrangement Prevailing for NASA Agena Satellite and Probe Applications

TYPICAL PROGRAMMING

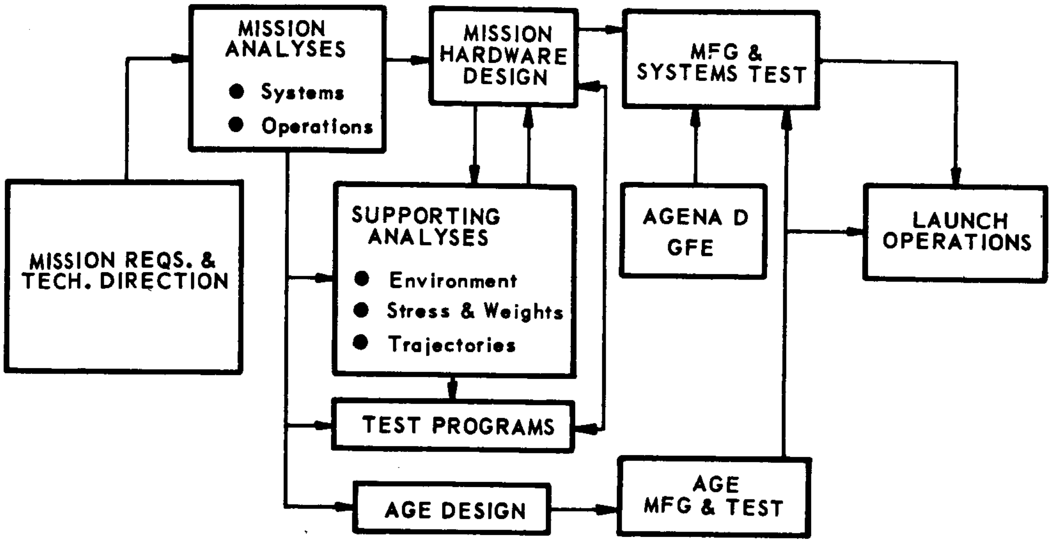

Figure 20-2 NASA Agena Program Activities Sequence

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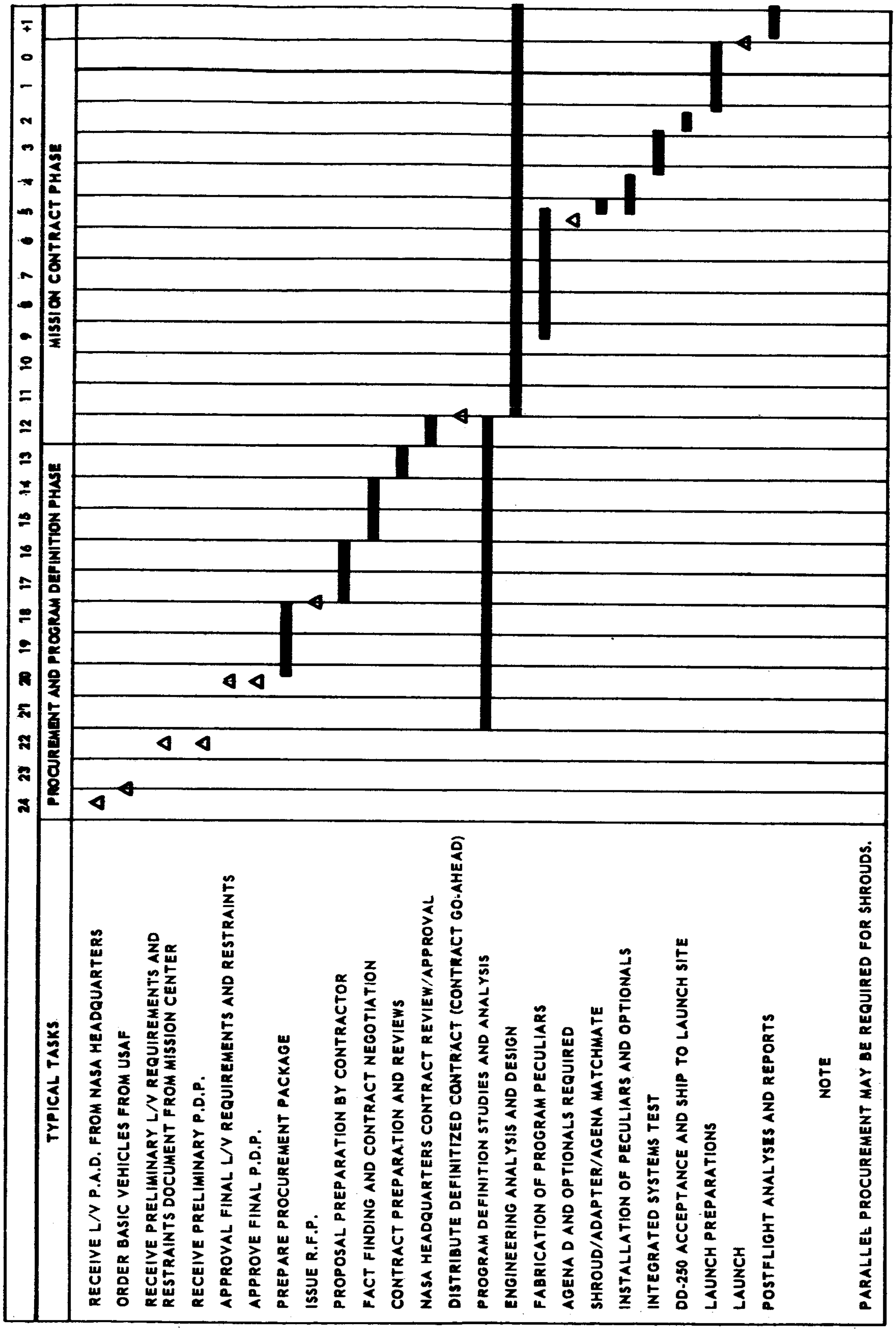

Fig. 20-3 Typical Procedure Cycle – Space Vehicle Integration and Agena Mission Adaptation

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20.2 VEHICLE HARDWARE PROGRAMMING

In the broadest terms, the typical LMSC program for a NASA Agena upper stage ascent booster application includes the following:

a. Adaptation of a GFE basic vehicle to the particular mission
b. Preliminary validation of the vehicle by an appropriate factory test program
c. Prelaunch validation and launch

Figure 20-4 diagrams the basic hardware sequencing from Agena D manufacturing and test through the completion of the final DD-250 acceptance of the mission vehicle, prior to shipment to the launch base.* Each program will deviate in certain respects from the sequence as diagrammed, but generally in an "additive" manner; that is, by adding activities in parallel, or interposing additonal tasks between the vehicle hardware program activities.

20.3 MISSION SOFTWARE PROGRAMMING

A typical LMSC program for an upper stage booster type application of the NASA Agena will include certain essential software or analysis activities at various levels of detail. The fundamental analytic services include the areas of aeromechanics, structures, thermodynamics, vehicle dynamics, and flight trajectory and performance — all of which generally support the hardware design and development program for the mission vehicle and the AGE. The capability in these basic analysis areas includes a large library of IBM 7094 and other computer programs which may be applied to the solution of specific problems.

After the initiation of a hardware program,** specific analytic studies may be required, either as preliminary to design or for validation of existing design in the particular application. The following studies are typical:

- Dynamic response of combined ascent vehicle, including mode shapes and vehicle control capabilities
- Ascent loads on entire vehicle due to winds and $\bar{\alpha}q$ conditions

---

*Similar diagrams detailing the terminal sequences at the two launch bases have been given in Section 19.

**Study efforts in advance of go-ahead may be carried out on a "Task" basis, as previously noted.

- Strength capability of booster interface, Agena, and shroud
- Thermal environment of spacecraft on pad and during ascent (shroud on and off); orbital environment of Agena
- Trajectory and performance: generation of preliminary, design, and final tag trajectories; range safety analyses; error analyses; allocation of propellant margins; sequence of events; and payload capability
- Free molecular drag, spin stability, separation analyses, and other studies of a specialized nature.

20.4 AGENA ASPECTS AFFECTING SPACECRAFT PROGRAMMING

Several Agena program events and activities have a direct bearing upon the spacecraft programming by levying requirements for initiative action, test participation, or hardware delivery, as discussed below.

20.4.1 Launch Vehicle Restraints Document

The restraints document must be prepared early in the program by the spacecraft contractor or agency. It should define the technical requirements and restraints imposed by the spacecraft upon the booster, booster adapter, Agena vehicle, shroud, spacecraft support structure, associated AGE, designated launch complex, and range. Details of these requirements are listed in the NASA Standard Requirements and Restraints Document.* Should preparation of the restraints document require LMSC inputs or supporting studies, LeRC has an "open-ended" studies contract arrangement with LMSC to perform such activities on a task basis in advance of a hardware commitment. It is also the responsibility of the Spacecraft Contractor or Agency to distribute on a scheduled basis reports on any changes or addendums to spacecraft mass, center-of-gravity, stiffness, moment of inertia, product of inertia, dynamic envelopes, mode shapes, etc.

*LMSC SP 3805-64-1 "Agena Missions Standard Requirements and Restraints Document, Vol. 1 - Format; Vol. 11 - Instructions"

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20.4.2 Interface Plan and Schedule Documentation (IPSD)

The IPSD will serve as the official plans and schedule document for all interface activities conducted during the spacecraft and launch vehicle design, test and manufacturing phases of a mission. Established interface activities conducted at the launch base are not to be included in the IPSD. Goddard Launch Operations at WTR and ETR will coordinate the interface activities at the launch bases using established procedures. The IPSD shall not contain information regarding the launch vehicle and AGE engineering evaluation test programs, qualification test programs, launch base tests, etc. This information is provided in the Integrated Test Plan. After coordination and concurrence of participating agencies, the publication and distribution of the IPSD is performed by the Launch Vehicle Systems Contractor.

Examples of those interface activities and events to be included in the IPSD are:

- a. Matchmates
- b. Hardware exchanges
- c. Special interface tests
- d. Interface document generation, review and releases for such documentation as:
  1. Requirements and Restraints Document
  2. Control drawings
  3. Program Requirements Document
  4. Launch Operations Plan
- e. Studies
- f. Design reviews
- g. Launch Base efforts that require definition early in the mission planning, or are unique in nature. Examples are:
  1. End-to-end calibration tests
  2. Combined Systems Test
  3. Special matchmates

20.4.3 Matchmates

20.4.3.1 Preliminary Matchmate. Preliminary matchmate tests are conducted jointly by LMSC and the spacecraft contractor during the design and development phase of the program. These test utilize dimensionally correct (not necessarily flight hardware) shroud systems, spacecraft adapters, and spacecraft to demonstrate mechanical compatibility at the various interfaces.

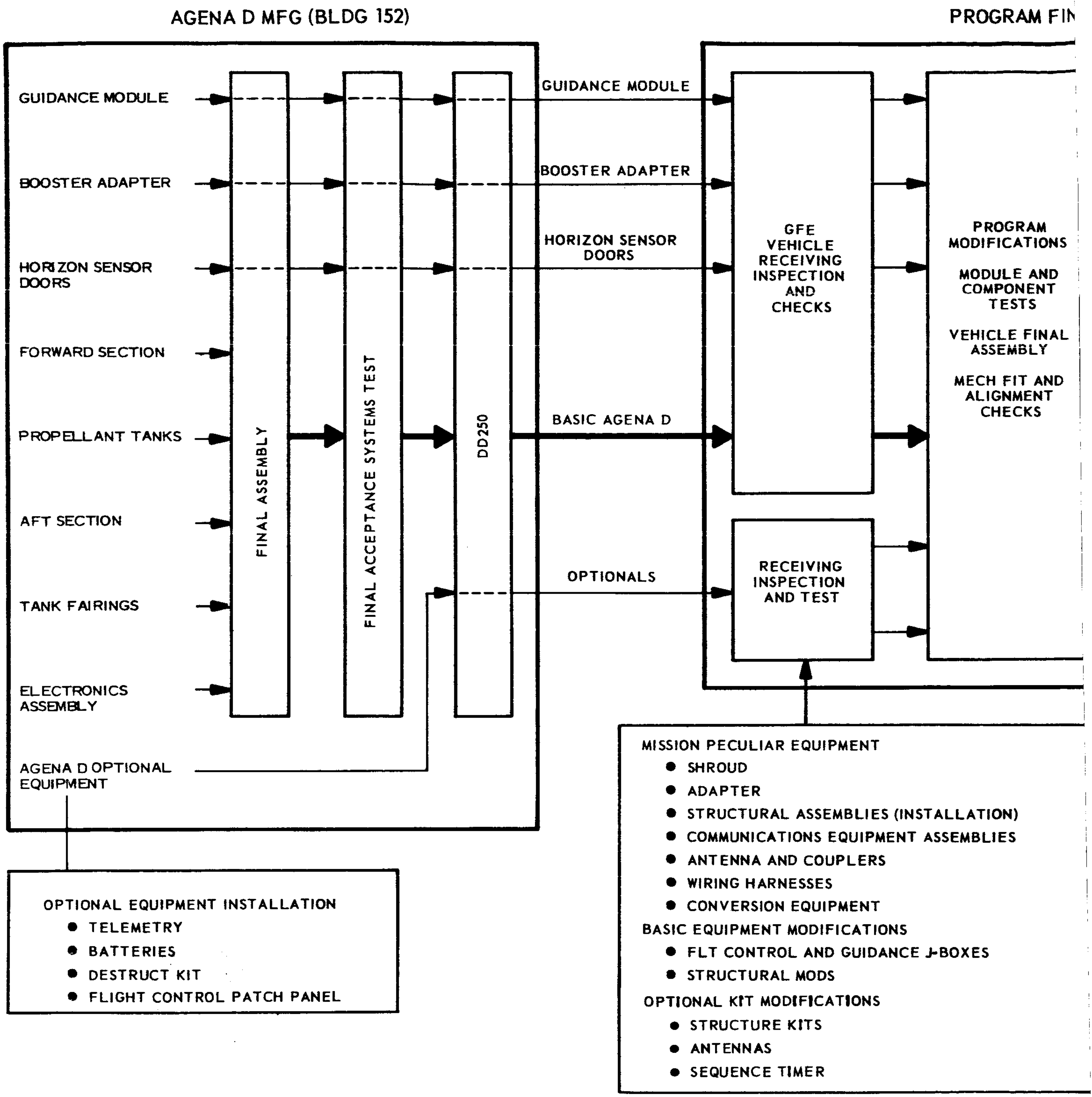
During these tests, the requirements for scraping or shimming to obtain minimum required contact and alignment at the interfaces is established, alignment of mating equipment such as electrical connectors is accomplished, shroud/spacecraft non-dynamic clearances are verified, and preliminary matchmate procedures are evaluated and modified as required. In addition, requirements for special tooling such as alignment fixtures, etc. (see AGE Section) are established. Spacecraft/Agena D electrical compatibility is generally verified during Agena D systems test through use of a spacecraft electrical simulator.

The matchmate procedures will vary with the hardware design; however, the scope and purpose will remain very much the same.

20.4.3.2 Final Matchmate. A final matchmate of flight hardware is performed either at LMSC or a spacecraft contractor facility. The following example, taken from LMSC-A374528A, Mariner D Integrated Test Plan, illustrates the purpose and scope of a typical final matchmate.

> Matchmate Tests
>
> Matchmate tests will be conducted at JPL on each flight spacecraft, flight shroud, and flight spacecraft adapter as a joint responsibility of JPL and LMSC. The tests will be conducted to establish mechanical and electrical compatibility of the spacecraft/spacecraft adapter/shroud systems. Certain RF checks and calibrations will also be performed to determine VSWR and insertion losses of the parasitic antenna and antenna coupler. Upon completion of the matchmate tests, the shrouds, spacecraft adapters, and spacecraft will be identified as sets and will remain so designated until launch unless unforeseen events prohibit this policy.

20.4.3.3 Electrical Compatibility Tests. In the case of Mariner Mars, electrical compatibility is checked both at Agena D Systems Test using a spacecraft simulator and at final matchmate using flight hardware.


AGENA D MFG (BLDG 152)
PROGRAM FIN
GUIDANCE MODULE
BOOSTER ADAPTER
HORIZON SENSOR DOORS
FORWARD SECTION
PROPELLANT TANKS
AFT SECTION
TANK FAIRINGS
ELECTRONICS ASSEMBLY
AGENA D OPTIONAL EQUIPMENT
FINAL ASSEMBLY
FINAL ACCEPTANCE SYSTEMS TEST
DD250
GUIDANCE MODULE
BOOSTER ADAPTER
HORIZON SENSOR DOORS
BASIC AGENA D
OPTIONALS
GFE VEHICLE RECEIVING INSPECTION AND CHECKS
RECEIVING INSPECTION AND TEST
PROGRAM MODIFICATIONS
MODULE AND COMPONENT TESTS
VEHICLE FINAL ASSEMBLY
MECH FIT AND ALIGNMENT CHECKS
OPTIONAL EQUIPMENT INSTALLATION
● TELEMETRY
● BATTERIES
● DESTRUCT KIT
● FLIGHT CONTROL PATCH PANEL
MISSION PECULIAR EQUIPMENT
● SHROUD
● ADAPTER
● STRUCTURAL ASSEMBLIES (INSTALLATION)
● COMMUNICATIONS EQUIPMENT ASSEMBLIES
● ANTENNA AND COUPLERS
● WIRING HARNESSES
● CONVERSION EQUIPMENT
BASIC EQUIPMENT MODIFICATIONS
● FLT CONTROL AND GUIDANCE J-BOXES
● STRUCTURAL MODS
OPTIONAL KIT MODIFICATIONS
● STRUCTURE KITS
● ANTENNAS
● SEQUENCE TIMER

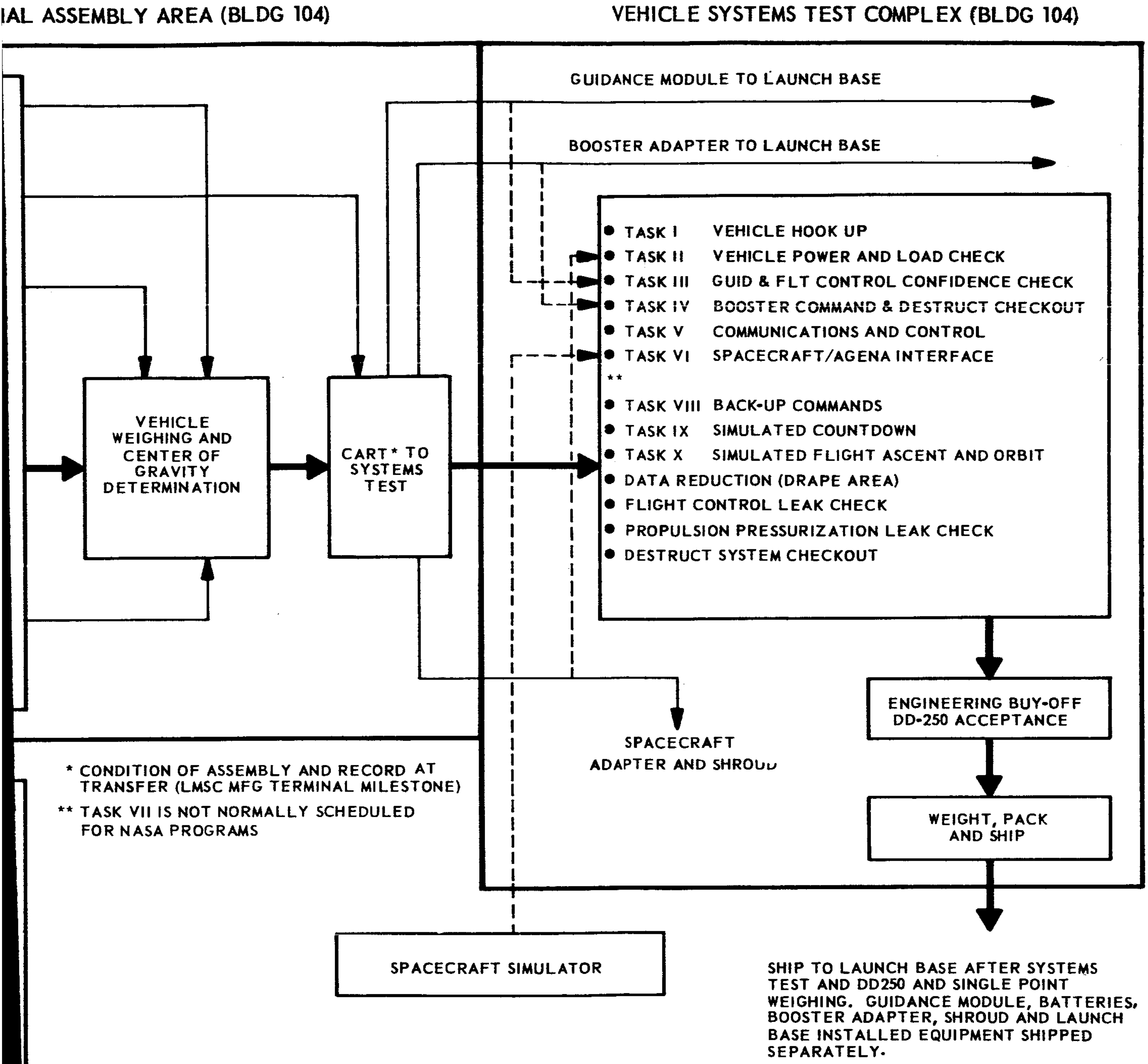


Figure 20-4 NASA Agena Factory Flow Chart

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20.4.4 NASA Agena Systems Test Requirements

20.4.4.1 Requirement for Spacecraft Simulator. LMSC performs a systems test on the program-configured flight Agena D prior to customer acceptance of the vehicle. The purpose of the systems test is to provide comprehensive verification of satisfactory performance of the Agena D vehicle on a systems level. The test is performed on an electrically mated Agena D, spacecraft adapter, spacecraft simulator, and booster adapter. A typical systems test includes the following:

1. Vehicle hookup test
2. Vehicle power and load tests
3. Guidance and flight control confidence checks
4. Booster command and destruct tests
5. Communications and control test
6. Guidance and control response
7. Systems test sequence of events backups
8. Simulated countdown test
9. Simulated flight test
10. Hydraulic fluid filtering and sampling
11. Flight control pneumatic system contamination and moisture test
12. Flight control pneumatic system functional and leakage test
13. Propulsion pressurization system leak check

As indicated above, a spacecraft electrical simulator is required during systems test to simulate the spacecraft electrical characteristics and loads imposed on the Agena D. A functional specification for the simulator as well as the simulator itself must be provided to LMSC by the spacecraft contractor in time to support the Agena D systems test.

20.4.3.2 Typical Spacecraft Simulator Example (Mariner D). The requirements for the Mariner Mars spacecraft electrical simulator are listed below to illustrate what a typical spacecraft contractor is expected to provide.

a. Purpose

The purpose of the simulator is to represent the Mariner Mars electrical circuitry and connects with the Agena D electrical circuits as shown in Fig. 20-5.

b. Design Characteristics

The simulator, electrical equivalent of the Mariner Mars, contains both active and passive circuits (Fig. 20-6). With the exception of the accelerometer, all these circuits are electrically connected to the Agena D by a three foot cable, terminating in a JPL separation connector.

Active Circuits

Data Encoder. The data encoder line signals will be square wave pulses at the rate of 600 pulses per second and will have an amplitude of five volts peak with a source impedance of approximately 750 ohms. The pulse rise time will be three microseconds at the source.

Accelerometer. The accelerometer circuitry which is not part of the spacecraft but rather that of the LMSC instrumentation of the Atlas/Agena/spacecraft combination will be represented by an attached cable (ENDEVCO 3030) of no less than three feet in length for attachment of the sensor.

Passive Circuits

Each passive circuit shall circulate at 10 milliamperes, 24v dc, through their respective passive loop and separation connector. There shall be an indication, at the simulator, for a successful test of each passive loop. The simulator's passive circuits are:

a. CC&S

b. Power subsystem

c. Tape Recorder

d. Plasma Probe

c. Power Requirements

The simulator's power supply requirement will be approximately 25 watts of 115v ac, 60 cps.

Operational Requirements

A mounting fixture to the Agena adapter plate will be furnished with prints and instructions for use of the simulator.

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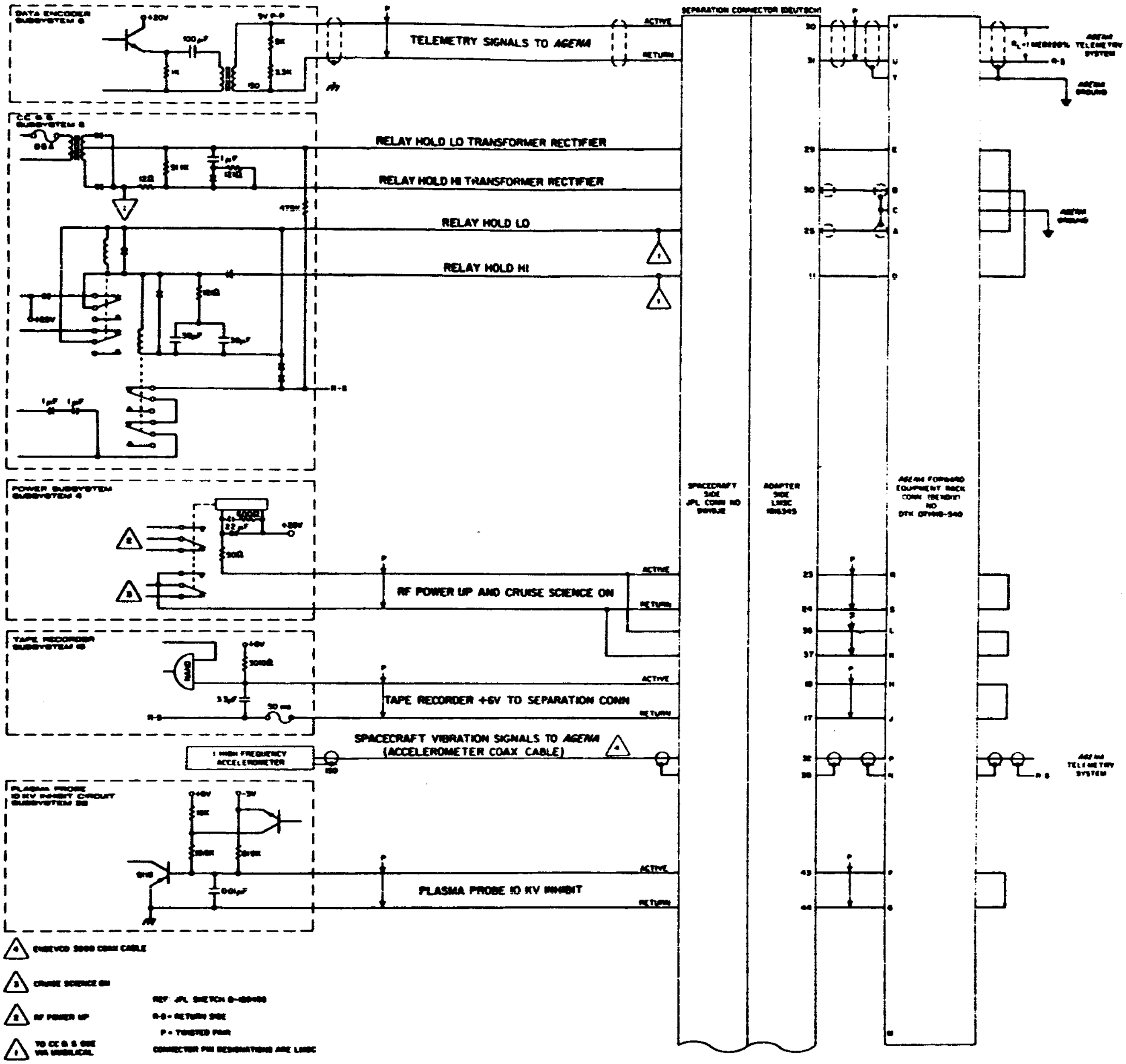

Figure 20-5 Agena/Spacecraft Circuitry (Mariner Mars) at Interface

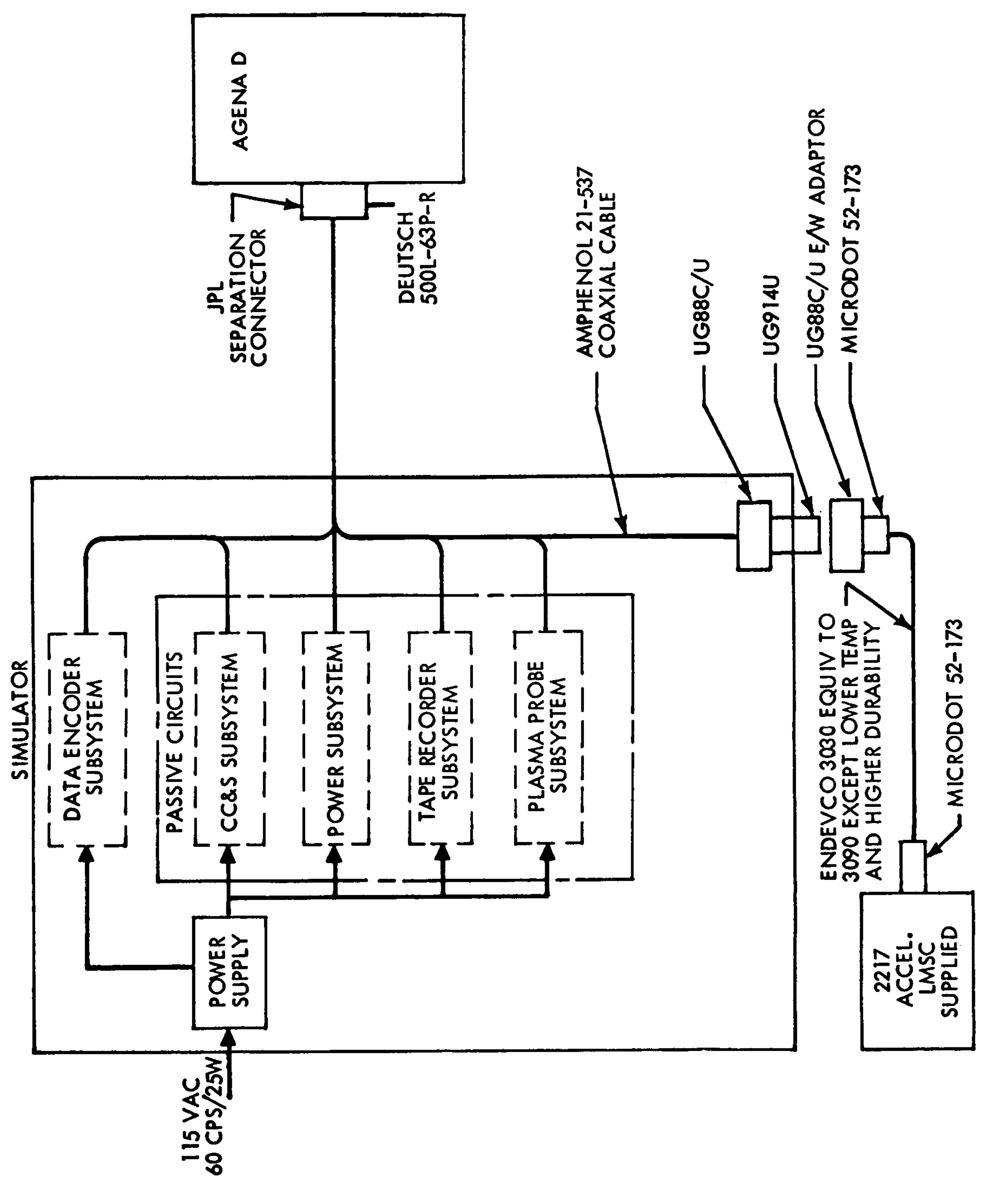

Figure 20-6 Agena/Spacecraft Electrical Simulator Circuitry at Interface (Mariner Mars)

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20.5 MISSION PECULIAR QUALIFICATION TEST PROGRAMS

Program peculiar Agena equipment such as the payload adapter may go through a series of qualification tests for the particular application if prior experience is not applicable. Many of these tests are of interest to the spacecraft designer either as a participant or as an observer. The test descriptions on several typical tests are listed below.

20.5.1 Modal Survey and Evaluation

A modal survey is conducted upon an assembly composed of a dynamic spacecraft model, and a complete shroud system mounted on the Agena interface structural members. The test assembly is rigidly mounted at the base in a vertical position with no restricting upper station supports. Several lower mode shapes and frequencies of the shroud are determined by the application of small oscillatory forces at an adequate upper station for each of the two lateral axes.

20.5.2 Spacecraft Separation System Qualification Tests

This test series demonstrates separation action, provides separation tip-off rates, and indicates effects of assymetric thrust and center of gravity (C.G.) location. The test assembly includes a spacecraft adapter, V-band clamp, simulated spacecraft, and separation system. The simulated spacecraft includes the total bus structure, possessing complete spacecraft inertial properties, and includes all equipment which applies external loads to the spacecraft. The following conditions are included in the test series:

a. Balanced spacecraft, nominal rate springs, misaligned separation mechanisms, and two bolts firing

b. Spacecraft C.G. offset, nominal spring rates, stroke and preload tolerances built into separation mechanisms, and two bolts firing

c. Balanced spacecraft, nominal spring rates, stroke and preload tolerances built into separation mechanisms, and two bolts firing

d. Spacecraft C.G. offset, max-min spring rates, stroke and preload tolerances built into separation mechanisms, and two bolts firing

e. Spacecraft C.G. offset, max-min spring rates, stroke and preload tolerances built into separation mechanisms, misaligned mechanism, and two bolts firing

f. Spacecraft C.G. offset, max-min spring rates, stroke and preload tolerances built into separation mechanisms, and one bolt firing

g. Spacecraft C.G. offset, max-min spring rates, stroke and preload tolerances built in, misaligned mechanisms, and one bolt firing

20.5.3 Shroud Pressurization Leak and Venting, Diaphragm Deflection, and Spacecraft Cooling Tests

Tests are conducted to establish the shroud system air flow and payload cooling characteristics, the adequacy of venting system to maintain and/or release pressure within prescribed limits, diaphragm deflections, and shroud system and adapter system leak rates.

The tests employ a complete shroud system in association with an Agena forward section, tank section fairings, and associated hardware; spacecraft adapter; simulated spacecraft; and AGE spacecraft cooling equipment. Agena aft equipment rack and the fairing exhaust configuration is simulated to provide equivalent exhaust pressure conditions for the following test objectives:

a. Suitability of the diaphragm vent system for ascent conditions and determination of the valve characteristics

b. Determination of diaphragm deflections in the aft direction when subjected to shroud pressurization, but no Agena forward section cooling air pressurization

c. Determination of diaphragm surge characteristics when subjected to shroud pressurization and intermittent Agena forward section cooling air pressurization

d. Determination of diaphragm deflections in the forward direction when subjected to Agena forward section cooling pressurization with no shroud pressurization

e. Evaluation of spacecraft cooling system characteristics and capabilities when subjected to spacecraft and ambient heat loads.

Spacecraft simulators will be required for the above and other LMSC development and qualification tests. These simulators will be required to impart the dynamic, structural, electrical, and thermal loads to the associated LMSC hardware depending upon the nature of the test. The characteristics may be incorporated in one or more spacecraft models depending upon the scheduled test utilization.

20.6 TYPICAL FLIGHT SUPPORT DOCUMENTATION

Flight support documentation is broken down into two categories: project required and range required. Figure 20-7 lists the project required documentation and the typical publication dates. The documentation for a WTR launch is generally the same as that for an ETR launch except that the Launch Test Directive is not used at WTR. Figure 20-8 lists the Range required documentation and the typical publication dates. Two differences in the documentation exists between the Ranges: ETR uses a booster requirements document; and WTR does not. The Flight Termination Report is required at L-6 months at ETR and L-2 months at WTR. If contract go-ahead is received after the date a document is normally required, the document is published as soon after contract go-ahead as possible. Typical interfunctioning of the flight support documentation is shown in Fig. 20-9. A brief description of the documentation is given below.

1. Project Development Plan (PDP). Describes the resources required to conduct the project and delineates the areas of overall responsibility among the agencies involved.
2. Planning Estimate (PE). Provides a general program description and summary of range support requirements for the program. The PE must be approved before detailed range planning on a test program can be initiated.
3. Program Requirements Document (PRD). Defines those program needs that are levied upon the launch support range by the user. The PRD constitutes the authorization for the range to take action to satisfy program needs.
4. Program Support Plan (PSP). Describes the proposed action to meet each requirement stated in the PRD. Lists program requirements to show: (1) which can be met from existing resources, (2) which can be met by programming for support capability, and (3) which cannot be met.

5. Operations Requirements (OR). Supplements and follows the PRD by describing in greater detail final test information, services, and related requirements for accomplishment of an individual test or test series within the overall program.

6. Operations Directive (OD). The range's response to the OR. It lists directives for mobilizing final resources to support requirements necessary for the test series.

7. Launch Operations Plan (LOP). The LOP directs the implementation of the entire flight test program. This plan presents specific objectives, systems description, data requirements and operating parameters to be used in support of the flight test program. (The LOP replaces the System Test Objectives document.) As part of the LOP, the Operations Package provides a check list to ensure final updating of documentation required for prelaunch and launch operations and a means of providing the updated information.

8. Flight Termination Report (FTR). Provides an overall description of the space vehicle destruct system, including wiring diagrams and photographs, and a summary of test results that will show capability of the system to perform its destruct functions adequately. The FTR is the basic instrument for obtaining approval of the flight termination system for use on the range.

9. Range Safety Report (RSR). Provides nominal trajectory data, dispersion data, and impact data resulting from malfunction or explosion during the ascent phase. The RSR provides basic data for obtaining range approval for a flight on a particular launch azimuth.

10. Pad Safety Report (PSR). Provides information on the pyrotechnics and ordnance items, the propellant and pressurization system, and the flight termination system associated with the program vehicle. The PSR establishes pad safety procedures for operations in the launch complex area.

11. Countdown Manual (CDM). Tabulates selected events from each of the individual agency countdowns to provide direction for the combined operations during the launch countdown. The CDM is also utilized by each agency for detailed launch countdown procedures for internal operations.

12. Launch Test Directive (LTD). The LTD is used for ETR launches only. The LTD provides direction to all launch base organizations for preflight test, checkout, and launch of the space vehicle.

13. Booster Requirements Document (BRD). The BRD is used at the ETR only. The BRD defines the standard Atlas/Agena range support requirements for a typical program. When applicable the program PRD may reference specific BRD pages.

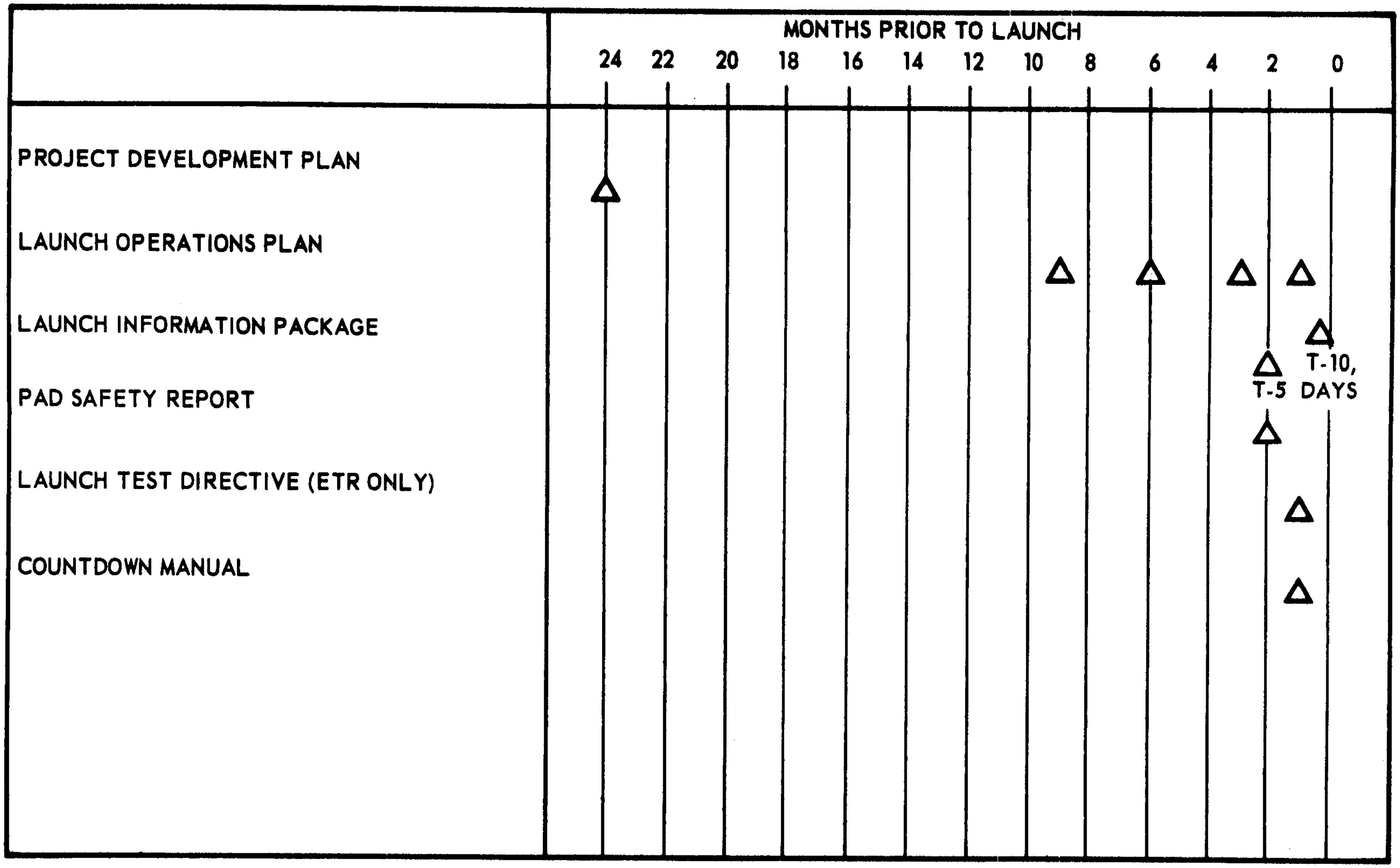

Fig. 20-7 Project Required Flight Support Documentation

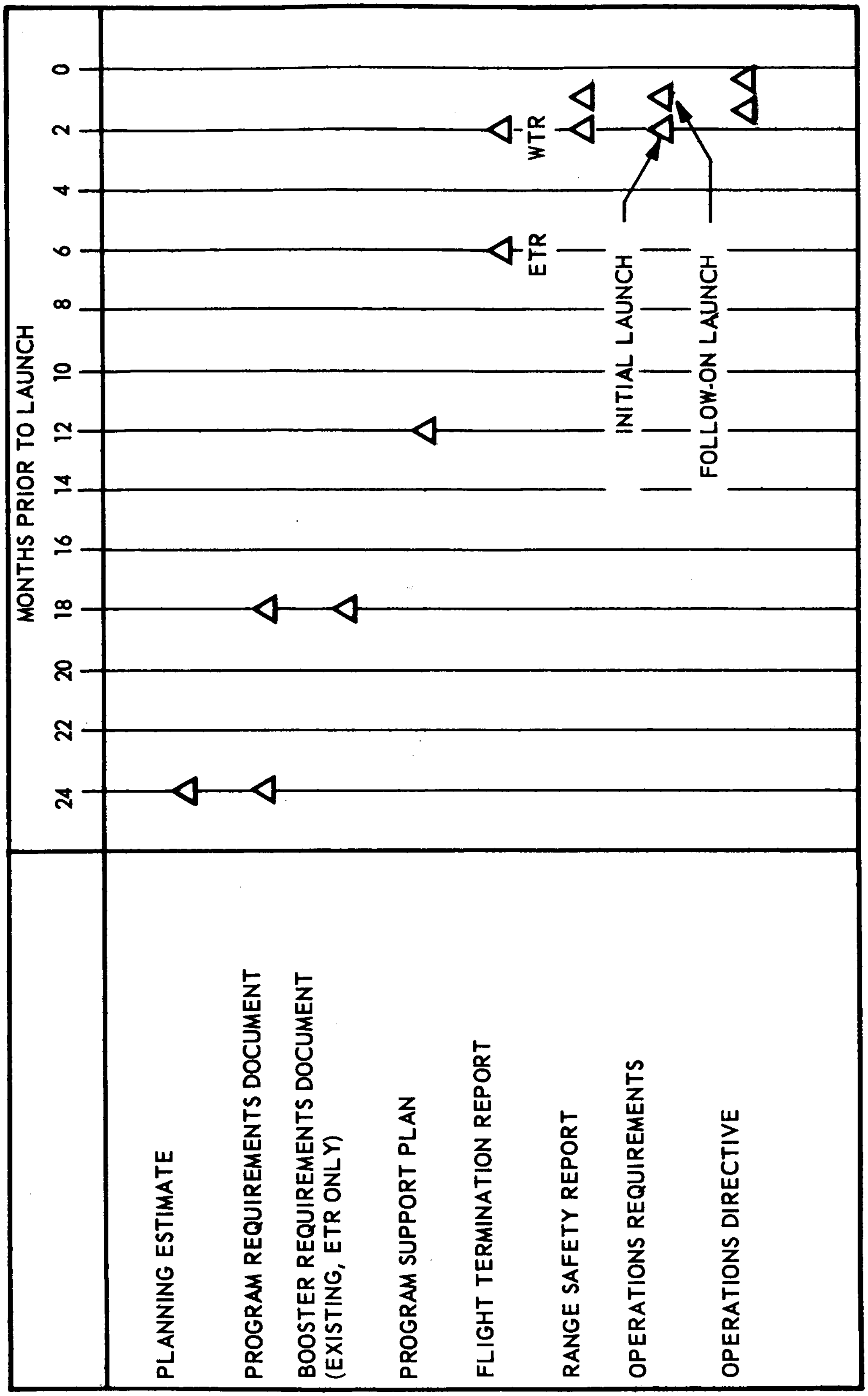

Fig. 20-8 Range Required Flight Support Documentation

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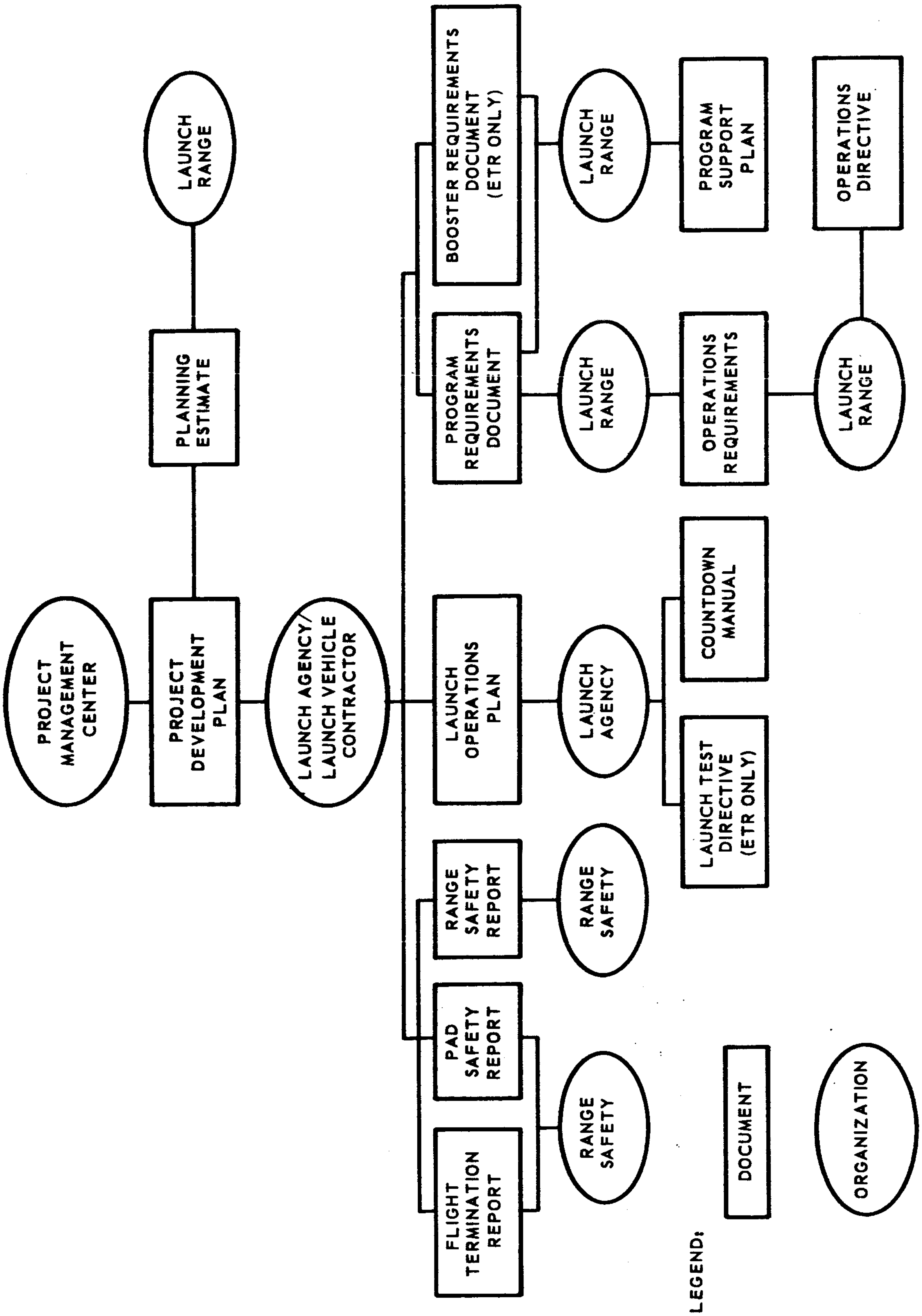

Fig. 20-9 Flight Support Documentation Chart

GLOSSARY

AF	Audio frequency
AFETR	Air Force Eastern Test Range (ETR)
AFWTR	Air Force Western Test Range (WTR)
AGE	Aerospace ground equipment
Agena B	Designation of dual-burn Agena used prior to Agena D, not used for procurements initiated after 1962.
Agena D	Standard vehicle used for all missions generally; also used to designate the vehicle as initially supplied GFE to the using programs and not including modifications.
A-12	Specific designation for spacecraft of Comsat-Echo passive communications satellite program, used synonymously with "Comsat" and "Echo" in reference to mission.
BAC	Bell Aerospace Corporation
Basic Equipment	Basic equipment, that which is required for most of the programs using the Agena D, consists of essential items of structure and equipment necessary to perform basic ascent, orbital or space functions. Basic equipment includes items that are required in the vehicle spaceframe, propulsion, electrical, guidance and control, and communications and electronics systems to achieve the common mission. For example, the rocket engine, propellant tanks, wire harnesses, and guidance module are items of basic equipment. A "Basic Agena D Vehicle" has everything required for an elementary ascent mission except for payload, nose fairing, batteries, beacon, and telemetry transmitter. Certain launch-base-installed basic equipment items such as the pyrotechnics and engine nozzle extension are transported directly to the launch base for installation in the Agena D vehicle during launch preparations.
BECO	Booster engine cutoff (Atlas)
BTL	Bell Telephone Laboratories

CART	Condition and record at transfer; an LMSC term used to describe the summary of the vehicle's condition at the time it is transferred from Manufacturing (usually to Systems Test)
C.G. or CG	Center of gravity
CMG	Control moment gyro
Comsat	Passive communications satellite (A-12) launched by a Thor/Agena from AFWTR, 25 January 1964
DAC	Douglas Aircraft Company
DD-250	Department of Defense Form number 250 used for accepting contractor completed equipment. This is the terminal milestone in factory sequence prior to shipment to launch base.
DM-21	Designation for Thor booster
Echo	Alternate designation for passive communications satellite, "Comsat"
EED	Electro-explosive device
EGO	Eccentric (orbiting) Geophysical Observatory, synonymously termed "EOGO"
EMI	Electromagnetic interference
ETR	Eastern Test Range (AFETR)
Fire	NASA project for investigating phenomena of reentry at hyperbolic (interplanetary) velocities
FM/FM	A multiplexed telemetry system which uses a subcarrier to frequency-modulate the RF carrier.
GAEC	Grumman Aircraft Engineering Corp
GC	Gas consumption
GD/A	General Dynamics, Astronautics
GE	General Electric
Gemini	NASA two-man rendezvous and docking program; intermediate manned spaceflight program between Projects Mercury and Apollo
GFE	Government furnished equipment

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GHE	Ground handling equipment
GLO	Goddard Launch Operations
GSE	Ground service equipment
GSFC	Goddard Space Flight Center
HIG	Hermetic integrating gyro
H/S	Horizon sensor
IR	Infrared
IRFNA	Inhibited, red, fuming nitric acid; oxidizer for Agena propulsion system
IRIG	Inter-range instrumentation group
IRP	Inertial reference package
ISIS-X	Ionospheric Sounding International Satellite – Experiment (Similar to S-27)
J-box	Junction-box for electrical connections
J-FACT	Joint flight acceptance composite test
JPL	Jet Propulsion Laboratory
LCS	Launch control system
LeRC	Lewis Research Center, NASA
LMSC	Lockheed Missiles & Space Company
LOB	Launch operations building
LOWG	Launch operations working group
LOX	Liquid oxygen
LPB	Launch pad building
LV-3A	Atlas SM-65D missile as adopted to space launch usage
MAB	Missile assembly building

Mariner C or Mariner Mars	Designation for Mars flyby program launched by an Atlas/Agena from AFETR on 28 November 1964
MDF	Mild detonating fuse
MECO	Main engine cutoff (Thor and TAT)
MFVO	Main fuel valve open
MIG	Miniature integrating gyro
MOPS	Mission operations (used in context of launch operations)
MSVP	Medium Space Vehicles Programs, NASA Agena Programs group of LMSC
NAA	North American Aviation
Nimbus	NASA meteorological satellite; missions launched by a Thor/Agena from WTR
OAO	Designation for Orbiting Astronomical Observatory, to be launched by an Atlas/Agena from ETR
OGO	Designation for Orbiting Geophysical Observatories (both spacecraft for EGO and POGO programs)
Optional Equipment	Items of equipment that perform specific functions beyond the capabilities of basic equipment, and which are used for more than one using program are available as optional equipment in kit form. Optionals are classified as "essential" and "selective." Essential optionals are those components that are required for all flights, such as batteries and flight control patch panels. Selective optionals are those components required for specific functions by the using programs such as the multi-start engine, command destruct, and the propellant dump kit. The kits include the wiring, bracketry, and plumbing necessary for installation and operation. Mounting provisions for installation are provided in the basic Agena D.
OTN	Over-the-nose (shroud)
PAM	Pulse amplitude modulation
PCM	Pulse code modulation
P/N	Part number
POGO	Polar Orbiting Geophysical Observatory; to be launched by a TAT/Agena from WTR

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POHCV	Pyro-operated helium control valve
PPM	Pulse position modulation
PPS	Primary propulsion system
Program Peculiar Equipment	Program peculiar equipment is that equipment other than basic or optional that is essential to perform the requirements of a particular program mission. Each program is expected to supply, as required, a program peculiar forward assembly consisting of appropriate nose fairing, fairing attach structure, payload, payload mating structure, and separation devices (if required). The program peculiar equipment supplied, developed, and qualified by the using program may also include special guidance or control equipment such as reaction wheels, secondary propulsion system, etc. In some instances, optional equipment may be reidentified as program peculiar for program identification such as program wired sequence timers and patch panels.
P-11	Designation for Agena subsatellite vehicle
Ranger	NASA program employing the Atlas/Agena launch vehicle for TV reconnaissance of the moon (hard lander).
RF	Radio frequency
RP-1	Rocket propellant 1 (standard hydrocarbon fuel similar to kerosene)
RVOC	Release valve off charge
S/C	Spacecraft
SECO	Sustainer engine cutoff (Atlas)
SLV-2A	Designation for TAT booster
SLV-3	Standard Launch Vehicle (Atlas)
SNAP	Space nuclear auxiliary power
SPS	Secondary propulsion system
SS/	Symbol for subsystem, followed immediately by the ldtter designation of the subsystem, i.e., SS/A – spaceframe (vehicle structure) SS/B – propulsion SS/C – electrical SS/C&C – communications and control SS/D – guidance

STA	Station, structural (usually measured in inches)
STL	Space Technology Laboratories
S-01B, SS-01B	Agena D vehicles
S-27	Mission designation for US-Canadian Topside Sounder project, launched 28 September 1962 from AFWTR by a Thor/Agena
TAT	Thrust-Augmented Thor
"tag"	Specific vehicle configuration data (as opposed to "batch" – value data) used for trajectory analysis
TAVE	Thor/Agena Vibration Experiment
TLM or T/M	Telemetry
Topside Sounder	Term used synonymously with "S-27" to designate the US-Canadian ionospheric sounding project
UDMH	Unsymmetrical dimethylhydrazine, fuel used for Agena propulsion system
UHF	Ultra high frequency
VAFB	Vandenberg Air Force Base, Air Force headquarters for AFWTR
VCO	Voltage-controlled oscillators
VECO	Vernier engine cutoff
VERLORT	Very long range tracking radar
VHF	Very high frequency
V/M	Velocity meter
VSWR	Voltage standing wave ratio
WTR	Western Test Range (AFWTR)
826	Designation for Air Force program using Agena

SP-3805-64-1-Vol I
1 July 1964

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AGENA MISSIONS STANDARD REQUIREMENTS AND RESTRAINTS DOCUMENT

(VOLUME I, FORMAT)

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Sheets

E-3236-2

REPORT CHANGE RECORD
FOR
AGENA MISSIONS STANDARD REQUIREMENTS AND RESTRAINTS DOCUMENT
(VOLUME I, FORMAT)

The following additions, revisions, or errata corrections, should be incorporated into the document identified above. This Report Change Record page should be inserted as the first page of the affected report preceding the title page. If a page in the original document is eliminated and/or replaced by the instructions which follow, the page must be destroyed.

ADDENDUM PAGE	REVISION		ERRATA		REVISION OR ERRATA CORRECTION (CORRECT IN INK)	CORRECTION MADE	
	REMOVE PAGE	INSERT PAGE	REMOVE PAGE	INSERT PAGE		INITIAL	DATE
			Title Page	New Title Page		[illegible]	9/18/64
			2-3	2-3			
			3-13	3-13			
			3-20	3-20			
			3-23	3-23			
			3-25	3-25			
			3-28	3-28			
			5-1	5-1			
			5-13	5-13			
	Report Change Record	Report Change Record					

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INTRODUCTION

Volume I, Format is to be used in conjunction with Volume II, Instructions, which presents detailed instructions for converting the format of Volume I to a mission requirements and restraints document for a specific program. The numbering system and identification letters of Volume II are keyed to those of Volume I.

All blank spaces in the format and all columns in the tables should be filled in at the time of issue of the document by inserting, as appropriate

a. The specific information required
b. "N/A" when information requested is not available
c. "Due," and the date of availability.

To facilitate handling where classified information is involved, it is recommended that the word "Classified" be placed in the appropriate blank spaces and the classified information be assembled and identified in a classified addendum.

E-3236-2

LAUNCH VEHICLE SYSTEM
REQUIREMENTS AND RESTRAINTS
FOR THE
________ PROGRAM

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CONTENTS

CONTENTS (Continued)

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CONTENTS (Continued)

ILLUSTRATIONS

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SECTION 1
GENERAL

1.1 SCOPE

This document defines the technical requirements and restraints imposed by the ____________________ spacecraft upon the booster, booster adapter, Agena D vehicle, shroud, spacecraft support structure, associated AGE, designated launch complex, and range.

1.2 PURPOSE

The purpose of this document is to set forth those technical requirements, as defined in the Scope, which are necessary to effect technical coordination between the various agencies involved in carrying out the ________________ Program. These agencies are:

a.
b. Lewis Research Center
c.
d. Lockheed Missiles & Space Company
e.
f.
g.

1.3 MISSION OBJECTIVES

1.4 DEFINITION OF SPACECRAFT SYSTEM

The spacecraft system interfaces and coordinates are defined by Figures 1-1 and 1-2.

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Figure 1-1 General Assembly of Spacecraft, Spacecraft Support Structure, and Shroud

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Figure 1-2 Coordinate System of Spacecraft and Launch Vehicle

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Figure 1-3 Plan View of Vehicle and Launch Pad, Showing Coordinate Axes

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SECTION 2
MISSION REQUIREMENTS AND RESTRAINTS

2.1 FLIGHT EVENTS

2.1.1 Spacecraft Flight Sequence of Events

Event Number (a)	Approximate Time From Liftoff (b)	Event Description (c)	Command Source Primary (d)	Command Source Backup (e)	Remarks (f)

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2.1.2 Maneuvers Required Prior to Spacecraft Separation

(See page 2-5 for table.)

2.2 OPERATIONS REQUIREMENTS

2.2.1 Launch Conditions

a. Range of launch azimuth and/or times

b. Launch dates

c. Launch periods

d. Other restraints

2.2.2 Trajectory Requirements

Spacecraft restraints on trajectory.

2.1.2 Maneuvers Prior to Spacecraft Separation

Maneuver Description (a)	Purpose (b)	Time		Velocity		An Requi	
		Requirement (c)	Tolerance (d)	Requirement (e)	Tolerance (f)	Roll	Pi

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Attitude													
gular (g) rement		Angular (h) Tolerance			Rate (i) Requirement			Rate (j) Tolerance			Special Restraints and Notes (k)
tch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw	

2

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A. Orbit Mission — Separable Spacecraft, Ascent Phase

1. Inclination angle
2. Orbit eccentricity
3. Orbit period or mean orbit altitude
4. Position of first perigee

    Reference Coordinate

5. Time at first perigee
6. Other controlled mission parameters:

    Parameter

    Value

B. Orbit Mission — Separable Spacecraft, Post-Ascent Phase

1. Inclination angle
2. Orbit eccentricity
3. Orbit period or mean orbit altitude
4. Position of first perigee

    Reference Coordinate

5. Time at first perigee
6. Other controlled mission parameters

    Parameter

    Value

C. Probe Mission — Separable Spacecraft, Ascent Phase

1. Parking orbit altitude
2. Time from Agena second-burn cutoff to target
3. Injection energy (vis viva)

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D. Probe Mission — Separable Spacecraft, Post-Ascent Phase

1. Target impact coordinates
2. Target impact velocities
3. Target orbit period or mean orbit altitude
4. Target orbit inclination
5. Target orbit eccentricity
6. Target orbit position of first perigee
7. Number of midcourse maneuvers
8. Time restraints on midcourse maneuvers
9. Velocity change required in midcourse maneuvers
10. Special Considerations

E. Reentry — Parameter

1. Velocity at Reentry Vehicle Separation
2. Velocity Azimuth
3. Flight Path Angle
4. Location of Reentry Vehicle Separation
   - Longitude
   - Latitude
   - Altitude
5. Impact Area
   - Longitude
   - Latitude

F. Special Trajectory Considerations

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2.3 TRACKING, COMMUNICATIONS, AND CONTROL REQUIREMENTS

	Areas of Coverage Required (a)	Classification (b)		Remarks (c)
		Mandatory	Desirable	
Telemetry				
Tracking				
Command Link				

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2.4 FLIGHT OPERATIONS AND COMMUNICATIONS

2.4.1 Data Acquisition System

2.4.2 Data Processing

2.4.3 Data Analysis

2.4.4 Decision and Command Operations

2.4.5 Ground Communications

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SECTION 3
DESIGN REQUIREMENTS AND RESTRAINTS

3.1 CONFIGURATION DEFINITIONS

3.1.1 Spacecraft

The general arrangement of the spacecraft is shown in Figure 3-1, which includes dimensions and gives locations and types of the principal equipment items, umbilical connections, and test plugs.

3.1.2 Spacecraft Support Structure

The general arrangement of the spacecraft support structure is shown in Figure 3-2, which includes principal dimensions, interface connections, and ducts.

3.1.3 Aerodynamic Shroud

The general arrangement of the shroud is shown in Figure 3-3, which includes principal dimensions, umbilical provisions, and access doors.

Figure 3-1 General Arrangement of Spacecraft

8 2

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Figure 3-2 General Arrangement of Spacecraft Support Structure

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Figure 3-3 General Arrangement of Aerodynamic Shroud

3.1.4 Composite Assembly

A drawing of both the spacecraft and shroud transition section is shown in Figure 3-4, which includes principal dimensions and interface connections.

3.2 MASS AND STIFFNESS PROPERTIES

3.2.1 Mass Properties

Mass Parameters	System Items: Spacecraft (Launch Condition) (a)	System Items: Spacecraft (Orbit Condition) (b)
Weight (lb)		
CG Location		
X		
Y		
Z		
Moments of inertia I_x		
I_y		
I_z		
Product of inertia I_{xy}		
I_{xx}		
I_{yz}		

Figure 3-4 Composite Assembly of Spacecraft and Shroud Transition Section

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A mass distribution diagram of the spacecraft is shown in Figure 3-5.

3.2.2 Stiffness Properties

Spacecraft structural stiffness factors as a function of station number are shown in Figure 3-6.

3.3 MECHANICAL INTERFACE REQUIREMENTS

3.3.1 Alignment and Tolerance

Spacecraft Axis (a)	Agena Axis (b)	Angle (c)	Tolerance (d)	Method of Defining Spacecraft Axes (e)

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Figure 3-5 Spacecraft Mass Distribution Diagram

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Figure 3-6 Spacecraft Structural Stiffness Factors

3.3.2 Matchmate

The matchmate tool shall be employed for the following purposes:

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3.4 ELECTRICAL INTERFACE REQUIREMENTS

3.4.1 Spacecraft/Spacecraft Support Structure

Electrical Interface

a. Number of interface connections and connectors:

b. Method of electrical bonding:

c. Electrical separation method:

d. Special considerations:

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3.4.2 Spacecraft Support Structure/Agena

a. Number of interface connections required by spacecraft:

b. Method of electrical connection:

c. Electrical separation method:

3.5 ELECTRICAL REQUIREMENTS

3.5.1 Instrumentation

(See page 3-15 for table.)

3.5.2 Spacecraft Power to be Supplied by Launch Vehicle

(See table on page 3- 17.)

3.5.1 Instrumentation

Type of Measurement (a)	Frequency Response or Sampling Rate (b)	Accuracy Required (c)	Range of Measurement (d)	T/M Coverage (e)

Purpose (f)	Classification		Location (i)
	Mandatory (g)	Desirable (h)	

2

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3.5.2 Spacecraft Power Supplied by Vehicle

(a)	(b) +28v Unregulated		(c) +28v Regulated		(d) -28v Regulated		115v 3ϕ 400 cps	
Spacecraft Function	Load	Duty Cycle	Load	Duty Cycle	Load	Duty Cycle	Load	Duty Cycle

1

(e)	115v 1ϕ 400 cps (f)			Other Power (g)									
							Regulation						
Power Factor	Load	Duty Cycle	Power Factor	Type	Volts	Freq	Volts	Freq	Phase	Load	Duty Cycle	Power Factor

2

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3.5.3 Switch Loading for Spacecraft Functions Activated Via Launch Vehicle

Event (a)	Switch Type (b)		Nature of Load (c)	Impedance (d)	Voltage (e)	Current (f)	
	Normally Open	Normally Closed				Peak	Steady

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3.6 ENVIRONMENTAL REQUIREMENTS

3.6.1 Spacecraft Thermal Environment

A. Spacecraft bulk temperature limits during:

1. Ground checkout shroud off

2. Transportation

3. Pad testing shroud on

The spacecraft power dissipation during testing periods is expressed as a function of time in Figure 3-7.

B. Maximum heat flux from the shroud inner surface to the spacecraft

C. Other requirements or restraints

D. Spacecraft component temperature limits

The temperature limits of the equipment listed in the table on page 3-23 shall not be exceeded.

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Figure 3-7 Spacecraft Power Dissipation During Testing Periods

3.6.1 Spacecraft Component Temperature Limits

Equipment (a)	Location (b)	Temperature Limits (c)		Power Dissipation (d)	
		High	Low	Average	Peak

Operating Intervals (e)	Spacecraft Equipment Subject to Radiation from Shield		Heat Capacity (h)	Basis of Requirement (i)
	Surface Absorptivity (f)	Surface Emissivity (g)		

2

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3.6.2 On-Pad Spacecraft Cooling

Shroud cavity circulation requirements:

Cooling media restrictions

Permissible pressure differential

Specific humidity limit

3.6.3 Contamination Control

Any contamination which occurs from ground cooling, pyrotechnic separation, or material outgassing shall be within the following limits:

A. Optical Degradation

At any critical location the transmittance and reflectance of an optical sample measured in the wave length of ___ to ___ microns shall be decreased no more than ____ percent.

B. Water Vapor

The water vapor content of cooling and purging gases shall not exceed _____ percent.

C. Contaminants

The contaminants contained in airconditioning, purging, or other gases introduced into the spacecraft area shall not exceed the limits in the following table (page 3-26).

3.6.3 Spacecraft Contamination

Location (a)	Particle Size (b) Metallic	Particle Size (b) Nonmetallic	Concentration (c)	Sampling Method (d)	Remarks (e)

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3.6.4 Sterilization

a. Vehicle areas to be sterilized:

b. Sterilizing medium or technique:

c. Time restraints:

d. Method of introducing sterilizing medium:

3.6.5 Applied Loads

The trajectory shall be so shaped that the following spacecraft design criteria shall not be exceeded.

3.6.6 Acoustic Noise

3.6.7 Electromagnetic Environment

A. Conducted Interference

Agena/Spacecraft Interface Conductors (a)	Noise Frequency or Transient Duration (b)	Voltage Level		Source Impedance (e)
		RMS (c)	PEAK (d)	

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B. EMI or RFI Tests

C. Radiated Interference

Equipment	Location	Radiation *	
		μ volts	Frequency
(a)	(b)	(c)	(d)

*Measurement in accordance with MIL-I-26600.

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D. Spacecraft Transmitter Identification (for transmitter operating prior to injection)

Make and Model (a)	Power		Antenna		Operating Frequency Range (f)
	Rated (b)	Meas'd (c)	Type (d)	Location (e)	

1

Bandwidth (g)	Time of Transmission (h)	Type of Modulation (i)	Multiplication System (j)

2

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E. Transmitter Frequency Spectra

Transmitter Make and Model (a)	Frequency (b)	Attenuation db Below Fundamental (c)

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F. <u>Spacecraft Wiring Design</u>

1. Ground system used

Single point	Location
Multiple point	Location
Other	Specify nature
Connection to Agena Ground	

2. Extent of conductor shielding used

Power	Exceptions
Signal	Exceptions
Pyrotechnics	Exceptions

3. Ground return system employed

   Signal

   Power

   Pyrotechnic

   Shield

   Equipment cases

4. Electrical bonding

   Method of bonding

5. Cable grouping

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G. Magnetic Materials

Special requirement for magnetic material usage at or near Agena/spacecraft interface:

a. Non-ferromagnetic materials:

b. Maximum allowable permeability of structural material (henry/meter).

H. Nuclear Radiation

Type of Radiation (a)	Energy (b)	Dose Rate (c)	Integrated Dose Rate (d)	Period of Integration (e)	Measured at (vehicle station) (f)	Contin-uous (g)	Inter-mittent (h)	Dura-tion (i)

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3.6.8 Miscellaneous Environmental Considerations

3.7 CLEARANCE REQUIREMENTS

3.7.1 Static Clearances

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3.7.2 Dynamic Clearances

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Figure 3-8 Aerodynamic Shroud Dynamic Envelope

3.8 SEPARATION REQUIREMENTS AND RESTRAINTS

3.8.1 Shroud

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3.8.2 Spacecraft Separation Parameters

Spacecraft Separation Parameters						Spin-Stabilized Spacecraft Separation			Remarks or Special Requirements
Angle and Tolerances			Angular Rate and Tolerances			Wobble Angle	Spin Rate	Angular Acceleration	
Roll	Pitch	Yaw	Roll	Pitch	Yaw	degrees	rad/sec	rad/sec	

SECTION 4
PROGRAM-PECULIAR TEST REQUIREMENTS

4.1 TEST REQUIREMENTS

4.1.1

4.2 STUDY REQUIREMENTS

SECTION 5
LAUNCH BASE REQUIREMENTS AND RESTRAINTS

5.1 TRANSPORTATION AND HANDLING CRITERIA

(See table on page 5-3.)

5.2 UMBILICAL AND TEST PLUGS

5.2.1 Electrical Umbilical

(See table on page 5-5.)

5.2.2 Electrical Test Plugs

(See table on page 5-7.)

5.2.3 Pressurized Gas Loading Umbilicals

(See table on page 5-9.)

5.2.4 Propellant Loading Umbilicals

(See table on page 5-11.)

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5.1 Transportation and Handling Criteria

Item (a)	Power During Transport (b)			Envr D
	Type	From	To	

...onmental Control ...ring Transport (c)	Special Requirements (d)

2

5.2.1 Electrical Umbilical

Item (a)	Function (b)	(c)			Shielding (d)			(e)
		Continuous	Duty Cycle	Size of Wire	Single	Twisted Pair	Group	Voltage Range

(f)	(g)	(h)				Classification (l)	
Current	Frequency	Impedance	Nature of Load (i)	Monitoring Area (Termination) (j)	Type of AGE Required (k)	Mandatory	Desirable

2

5.2.2 Electrical Test Plugs

Item (a)	Function (b)	(c)			Shielding (d)			(e)
		Continuous	Duty Cycle	Size of Wire	Single	Twisted Pair	Group	Voltage Range

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(f)	(g)	(h)				Classification (l)	
Current	Frequency	Impedance	Nature of Load (i)	Monitoring Area (Termination) (j)	Type of AGE Required (k)	Mandatory	Desirable

2

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5.2.3 Pressurized Gas Loading Umbilicals

Type of Gas (a)	Specification (b)	Prelaunch Max. Allowable (c)	
		Press.	Temp.

Launch Pad (d)		Total Volume at Std. Temp. & Press. (e)	Dumping and Venting Requirements (f)
Press.	Temp.		

2

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5.2.4 Propellant Loading Umbilicals

Type of Propellant (a)	Loading Flow Rate (b)	Pressure (c)	

Temperature (d)	Total Mass (e)	Dumping and Venting Requirements (f)

2

5.2.5 Parasitic Coupler and Reradiating Antennas

(See table on page 5-15.)

5.3 LAUNCH BASE SEQUENCING

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5.3.1 Spacecraft Assembly Building Operations

(See table on page 5-17.)

5.3.2 Pad Checkout (Gantry in Place)

(See table on page 5-19.)

5.2.5 Parasitic Coupler and Reradiating Antennas

Function (a)	Location (b)	Allowable Attenuation (c)	Stability (d)	Repeat-ability (e)

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5.3.1 Spacecraft Assembly Building Operations

Spacecraft Operation (a)	Schedule Days Prior to Launch (b)		Support Equipment (c)	
	Start	Stop	Item	Support

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Description of Support (d)	Restraints and Validation Time Period (e)

2

5.3.2 Pad Checkout (Gantry in Place)

Spacecraft Test or Operation (a)	Time Periods (b)		Support Equipme[nt] (c)	
	Duration	Time From Launch	Equipment Required	E

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nt		
quipment Supplier	Test or Operation Requirements (d)	Communication Requirements (e)

2

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5.3.3 Countdown Activity

		Time Prior to Launch
a. RF Transmission		______
b. Spacecraft Fueling and Topping		______
c. Switch to Internal Power		______
d. Others	(1) ______	______
	(2) ______	______
	(3) ______	______
	(4) ______	______

5.3.4 Pad Cabling Requirements

Connections	Number of Conductors	Wire Size	Shielding
(a) Spacecraft to Launch Pad Bldg.			
(b) Spacecraft to Launch Operations Bldg.			
(c) Launch Pad Bldg. to Launch Operations Bldg.			
(d) Other (1) ______			
(2) ______			
(3) ______			

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REVISIONS AND UPDATING

This document will be revised as necessary to maintain it in an accurate, up-to-date status. When such revisions are made, revised or addendum pages will be provided to holders of this document. Each such revised or addendum page will be marked in the upper right hand corner with the date of issue, and each issue will be accompanied by a Report Change Record indicating affected pages and instructing the recipient. The Report Change Record format is shown on the following page.

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REPORT CHANGE RECORD FOR AGENA MISSIONS STANDARD REQUIREMENTS AND RESTRAINTS DOCUMENT (VOLUME I, FORMAT)

The following additions, revisions, or errata corrections, should be incorporated into the document identified above. This Report Change Record page should be inserted as the first page of the affected report preceding the title page. If a page in the original document is eliminated and/or replaced by the instructions which follow, the page must be destroyed.

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ADDENDUM PAGE	REVISION		ERRATA		REVISION OR ERRATA CORRECTION (CORRECT IN INK)	CORRECTION MADE	
	REMOVE PAGE	INSERT PAGE	REMOVE PAGE	INSERT PAGE		INITIAL	DATE

SP-3805-64-1-Vol II
7 October 1964

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AGENA MISSIONS STANDARD REQUIREMENTS AND RESTRAINTS DOCUMENT

(VOLUME II, INSTRUCTIONS)

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INTRODUCTION

Volume II, Instructions is to be used in conjunction with Volume I, Format. It presents detailed instructions for converting the format of Volume I to a mission requirements and restraints document for a specific program. The numbering system and identification letters of Volume II are keyed to those of Volume I.

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Title Page: Insert the official name of the program together with the abbreviated form or designation.

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CONTENTS

CONTENTS (Continued)

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CONTENTS (Continued)

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ILLUSTRATIONS

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GLOSSARY

AG	Product of Area and Shear Modulus
BECO	Booster Engine Cutoff
Bonding	Electric Wire Connection Between Surfaces
cps	Cycles per second
DAC	Douglas Aircraft Company
db	Decibel
EI	Product of Young's Modulus and Moment of Inertia
EMI	Electro-Magnetic Interference
FM/FM	Frequency Modulated Sub-Carriers on Frequency Modulated Carrier
GD/A	General Dynamics/Astronautics
I_x	Moment of Inertia about the Roll Axis
I_{xx}	Product of Inertia in XX plane
I_{xy}	Product of Inertia in XY plane
I_y	Moment of Inertia about the Pitch Axis
I_{yz}	Product of Inertia in YZ Plane
I_z	Moment of Inertia about the Yaw Axis
JG	Product of Polar Moment of Inertia and Shear Modulus
kMc	Kilo-Megacycles
LeRC	Lewis Research Center
LMSC	Lockheed Missiles & Space Company
PAM	Pulse Amplitude Modulated
RF	Radio Frequency
RFI	Radio Frequency Interference
RMS	Root Mean Square
Vis Viva	Total energy $(1/2\ V^2 - \frac{\mu}{r})$

X	Roll Axis
Y	Pitch Axis
Z	Yaw Axis
α	Nucleus of Helium Atom Ejected in Certain Radioactive Disintegration
β	Electron Emitted from Radioactive Disintegration
γ	Radiation Rays Emitted During Nuclear Transformations

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SECTION 1
GENERAL

1.1 SCOPE

Insert the official name of the program together with the abbreviated form or designation.

1.2 PURPOSE

Insert the names of:

Item (a):	The spacecraft center
Item (b):	The launch system management center--LERC
Item (c):	The booster system contractor (If Atlas, insert GD/A) (If TAT, insert DAC)
Item (d):	The Agena D and Agena D system contractor--LMSC
Item (e):	The spacecraft system contractor

1.3 MISSION OBJECTIVES

Provide a general description of the launch vehicle, range required, mission trajectory, and mission end objectives that the spacecraft is designed to satisfy.

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1.4 DEFINITION OF SPACECRAFT SYSTEM

Figure 1-1: Illustrate by two or more views the general assembly of the spacecraft, spacecraft support structure, and aerodynamic shroud. The views should provide approximate profiles of the system components and show interfaces and major dimensions. (See sample illustration.)

Figure 1-2: Provide the coordinate systems for the spacecraft, booster, and Agena D vehicle. Details of the coordinate systems of the booster and the Agena D are given in Ref. 1. (See sample illustrations, Figures 1-2a and 1-2b.)

Figure 1-3: For the vehicle mounted on the launch pad, show spacecraft coordinate axes in relationship to the Agena D reference system and true north. Details are given in Ref. 1. (See sample illustrations, Figures 1-3a and 1-3b.)

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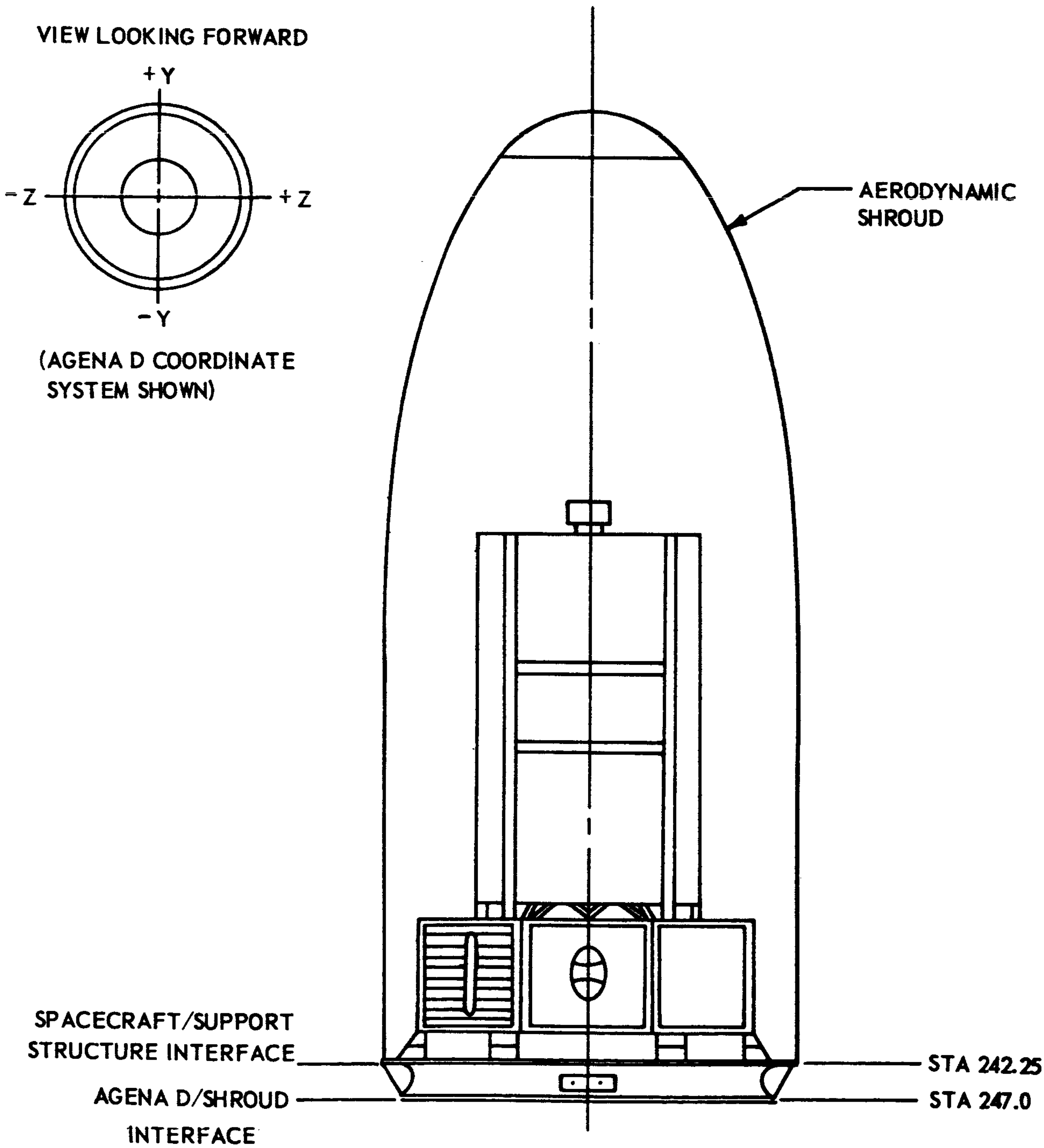

Figure 1-1 General Assembly of Spacecraft, Spacecraft Support Structure, and Shroud

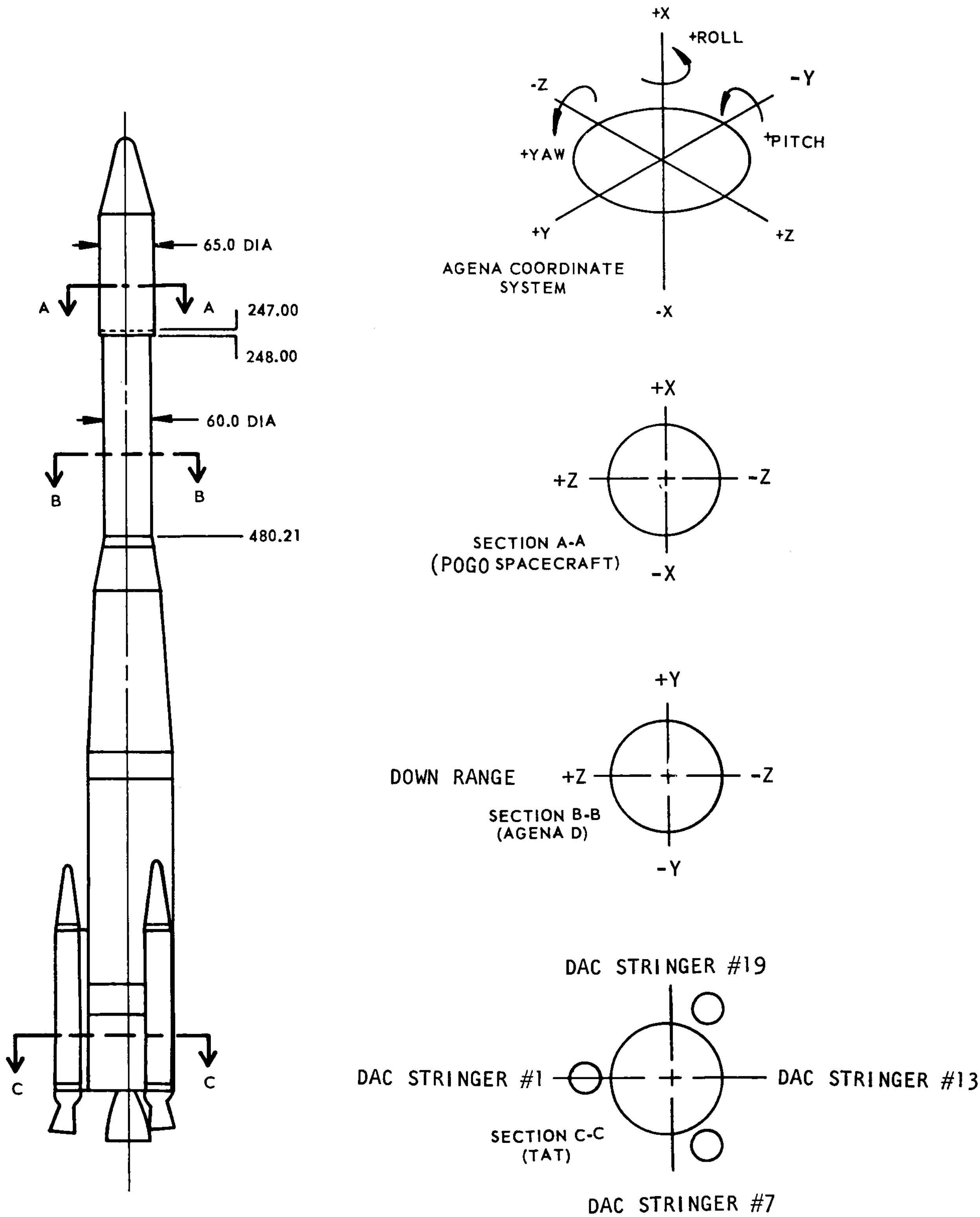

Figure 1-2a Coordinate System of Spacecraft and Launch Vehicle (TAT Booster)

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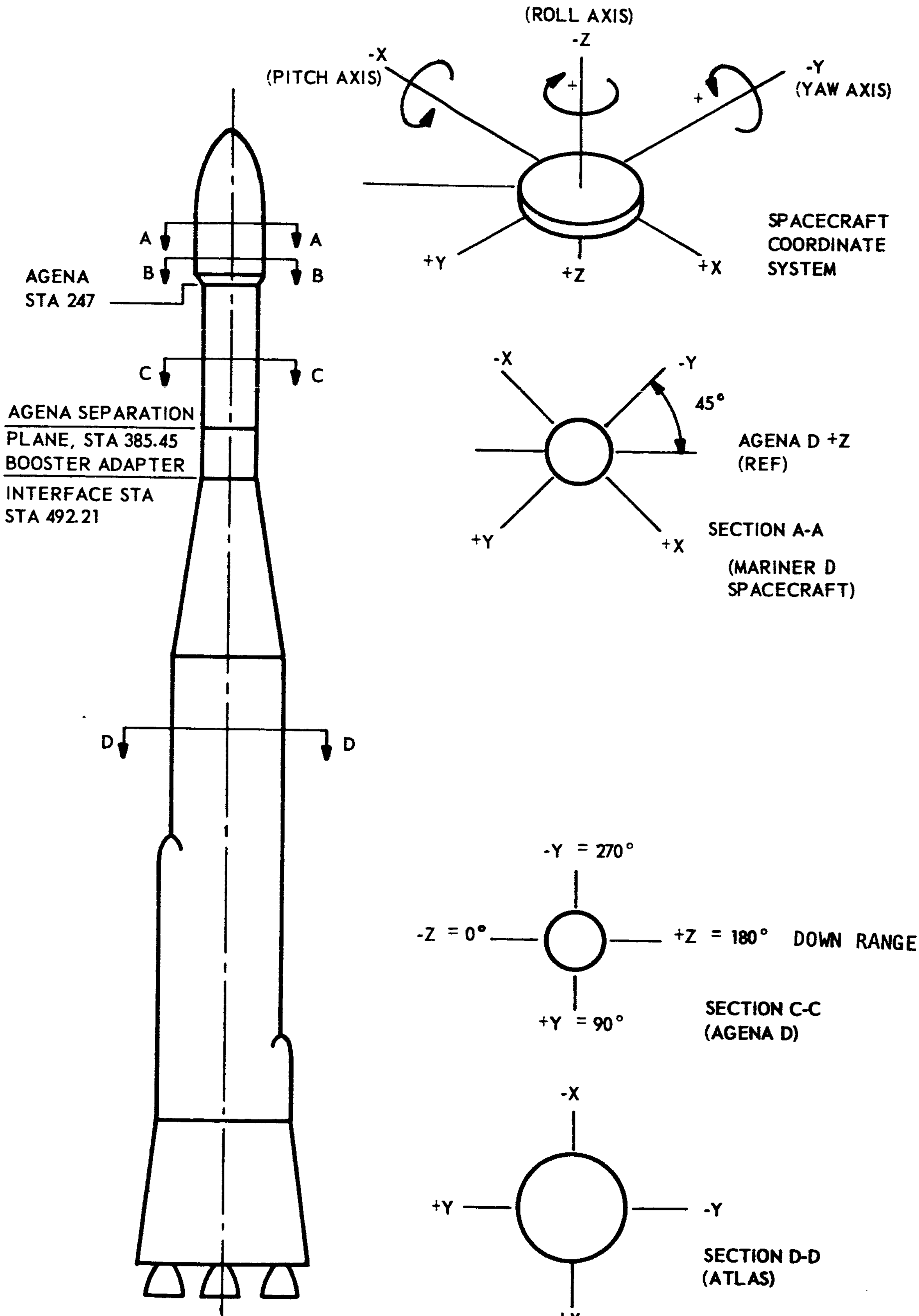

Figure 1-2b Coordinate System of Spacecraft and Launch Vehicle (Atlas Booster)

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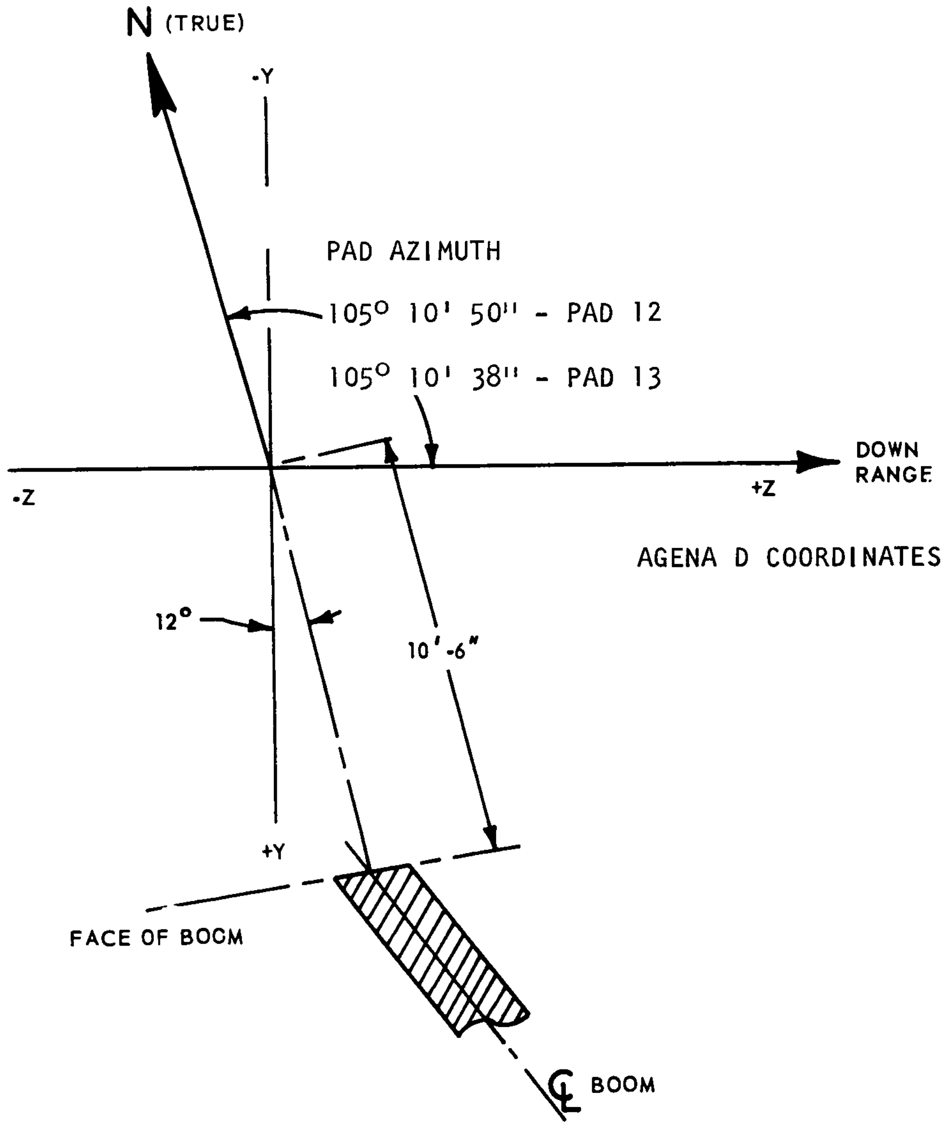

Figure 1-3a AMR Pads 12 and 13
Showing Coordinate Axes

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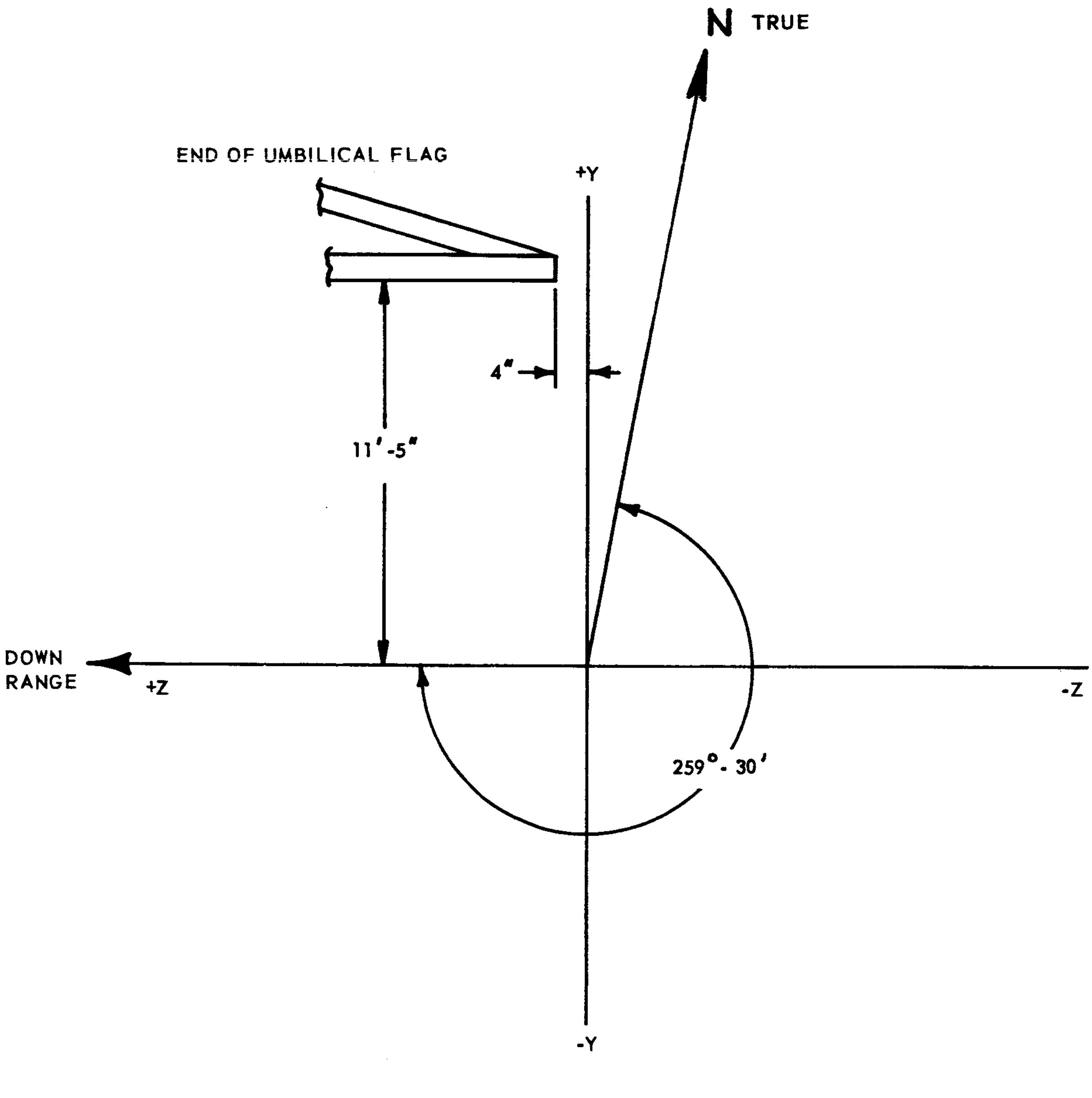

Figure 1-3b PMR Complex 75-1, Pad 1, Showing Coordinate Axes

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SECTION 2
MISSION REQUIREMENTS AND RESTRAINTS

2.1 FLIGHT EVENTS

Provide a general description of mission trajectory, altitude requirements, arrival times, and other restraints that cover all pertinent details of the spacecraft flight events.

2.1.1 Spacecraft Flight Sequence of Events

The sequence to be defined shall consist of programmed items which affect, first, the spacecraft/Agena electrical interface, and, second, the operations of spacecraft mechanisms that could interfere with operation of the Agena. Examples of the first are commanded separation by both Agena and spacecraft programmers, activation of spacecraft power supply by Agena programmer, and special Agena maneuvers. Examples of the second are start-up of spacecraft rotating machinery, initiation of spacecraft RF emission, and initiation of spacecraft nuclear power equipment.

Column (a): Insert event-identifying numbers.

Column (b): Insert approximate time of event from liftoff or other significant mission event such as Agena second burn and identify the mission event used.

Column (c): Describe the event to be accomplished.

Column (d): Specify the primary source of the command that initiates the event. The primary source is defined as that which operates on the nominal mission in the absence of malfunction.

Column (e): Specify the backup command source; i.e., the command device which operates in the event of a malfunction in the primary device.

Column (f): Provide the reason for the event. Also note any event which is time restrained to any other event.

2.1.2 Maneuvers Required Prior to Spacecraft Separation

This section shall define all maneuvers required prior to spacecraft separation. Maneuver limits, pitch accuracies, rates, etc., are given in Ref. 1.

Column (a): Describe the maneuver by the action required.
Column (b): Indicate reason for performing the maneuver.
Column (c): Indicate approximate time of maneuver with respect to liftoff or other significant mission event.
Column (d): Specify any restrictions on the time allowed to complete each maneuver.
Column (e): Specify the velocity change required in the maneuver.
Column (f): Specify the variation permitted in the velocity change.
Column (g): Specify the attitude requirement during the maneuver. (Ref. Para. 3.8.2 for definition of angles.)
Column (h): Specify attitude tolerances allowed.
Column (i): Specify the attitude rate requirement.
Column (j): Specify the rate tolerances allowed.
Column (k): Specify all special requirements and include any pertinent information, e.g. precautions to be taken to ensure that spacecraft-Agena collision does not occur.

2.2 OPERATIONS REQUIREMENTS

2.2.1 Launch Conditions

Item (a): Indicate the launch azimuth sector (maximum and minimum launch azimuth) or the launch time interval (within the launch date) that may be utilized for this mission.

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Item (b): Indicate the target launch dates for the mission.

Item (c): Indicate the permissible first and last launch dates for each vehicle.

Item (d): Indicate any other restraints that limit and affect the launch.

NOTE: Launch dates should be in unclassified form.

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2.2.2 Trajectory Requirements

Indicate any spacecraft-peculiar requirements that would restrict the development of the ascent trajectory, e.g., radioactive payload restraints on range safety. (See also Ref. 2.)

A. Orbit Mission - Separable Spacecraft, Ascent Phase

Items 1 thru 4: Indicate mission requirements for the orbit injection parameters and the tolerable variation in each parameter.

Items 5 and 6: Indicate the reference with respect to which the position of first perigee is measured, e.g., first ascending node.

B. Orbit Mission - Separable Spacecraft, Post-Ascent Phase

Items 1 thru 6: Complete this section as in 2.2.2A for those missions wherein the final orbit is changed from that of the injection orbit. Repeat this section as many times as is necessary to define each orbit change.

C. Probe Mission - Separable Spacecraft, Ascent Phase

Item 1: Enter the parking orbit altitude to be used (circular orbit assumed).

Item 2: Indicate planned transfer time from Agena second burn cutoff to target. Target is defined as the planetary body or trajectory end condition for the mission.

Item 3: Indicate the Vis-Viva energy integral value in km^2/sec^2. Specify minimum and maximum values. Use curves if necessary.

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D. Probe Mission — Separable Spacecraft, Post-Ascent Phase

Item 1: Provide impact coordinates on target planetary body or trajectory end point and the permissible variation in these coordinates.

Item 2: Indicate impact velocities and variation.

Items 3 thru 6: Indicate the parameters and variations listed when the mission requirement involves establishment of an orbit about another planetary body.

Items 7 thru 9: Indicate restraints on accomplishing the mid-course velocity corrections including considerations of tracking, trajectory computation, and look angles on the time of correction accomplishment. Indicate the range of mid-course correction and total velocity change capability required.

Item 10: Indicate restraints or special requirements.

E. Reentry

Items 1 thru 5: Indicate those parameter values and permissible variations that are pertinent to a mission terminating in reentry of the Earth's atmosphere. Indicate priority of parameters with respect to accuracy (greatest accuracy is number one, etc.).

F. Special Trajectory Considerations

Indicate any other requirements that affect the trajectory, e.g., requirements imposed by radar coverage limits, etc. (Ref. 2).

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2.3 TRACKING, COMMUNICATION, AND CONTROL REQUIREMENTS

In this section, the requirements for acquisition of telemetry data, tracking of the booster or Agena for spacecraft purposes and the accomplishment of command communication shall be specified. Tracking and/or telemetry acquisition is defined by those time periods when the vehicle is at least five degrees above the local tangent plane at the ground station.

Column (a): Indicate those geographic areas over which one or more of the functions of tracking, communication, or control shall be accomplished. Alternatively, identify the area of coverage by the corresponding mission flight phase, e.g., launch to BECO, Agena first or second burn, spacecraft separation, Agena retro, etc.

Column (b): Indicate the classification of the required functions as either mandatory or desirable. Indicate under "Desirable" the relative priority value for each function by assigning the most important function a value of 1 and indicating functions of lesser priority by successively higher numbers.

Column (c): Indicate reason for requirement.

2.4 SPACECRAFT FLIGHT OPERATIONS AND COMMUNICATIONS

2.4.1 Data Acquisition System

Specify where and by which agency or contractor the flight data is acquired for use in the real-time decision and command operations.

2.4.2 Data Processing

Describe the equipment required for flight data processing for use in the real-time decision and command operations. Specify where the equipment is required and by whom.

2.4.3 Data Analysis

Describe the equipment required for flight data analysis for use in the real-time decision and command operations. Specify where the equipment is required and by whom.

2.4.4 Decision and Command Operations

Describe location, equipment, and support required from each contractor for decision and command generation. Describe equipment for command transmission to the spacecraft.

2.4.5 Ground Communications

Describe communication link and information flow between the above functions.

3.1 CONFIGURATION DEFINITIONS

3.1.1 Spacecraft

Figure 3-1: Provide a general arrangement drawing of the spacecraft. Indicate overall dimensions, locations of major equipment items, and umbilical connections. (See sample illustration.)

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3.1.2 Spacecraft Support Structure

Figure 3-2: Provide a general arrangement drawing of the spacecraft support structure including sealing diaphragm and attitude spin stabilization system, if applicable. Indicate overall dimensions, locations of interface connections and ducts, and RF transparency requirements. (See sample illustration.) Details of standard equipment are available in Ref. 1.

3.1.3 Aerodynamic Shroud

Figure 3-3: Provide a general arrangement drawing of the aerodynamic shroud. Indicate overall dimensions, locations of required umbilical connections if different from standard, and RF transparency requirements. Details of standard equipment are available in Ref. 1. If a special shroud is required, state which characteristics are needed but are not provided by a standard shroud. (See sample illustration.)

3.1.4 Composite Assembly

Figure 3-4: Provide an assembly drawing of the spacecraft and shroud transition section. (See sample illustration.)

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SECTION 3

DESIGN REQUIREMENTS AND RESTRAINTS

Interface documentation will be prepared by an organization responsible to LeRC to describe in detail all physical and functional requirements of interfaces between systems and major assemblies designed or provided by associate contractors and the systems contractor. The documentation will completely specify all critical characteristics of mechanical, electrical, hydraulic, pneumatic, optical, RF, and weight requirements at the interface. Interface documentation shall show:

(a) Design details which are essential to assuring compatibility of the affected systems or assemblies.

(b) Reference documentation which is required to support systems analyses such as applicable drawings, specifications and analysis report.

Review and approval of these documents by affected associate contractors, the organization responsible to LeRC, LeRC and other affected government agencies, as applicable, will assure that all interface agreements and directives are properly documented.

Requests for revisions by program participants will be forwarded in writing to LeRC. Any necessary changes will be implemented by the organization responsible to LeRC in accordance with procedures established for the program.

Periodic meetings will be held among LeRC, the other government agencies involved, the organization responsible to LeRC, and associate contractors to ensure proper maintenance of the interfaces and related documentation.

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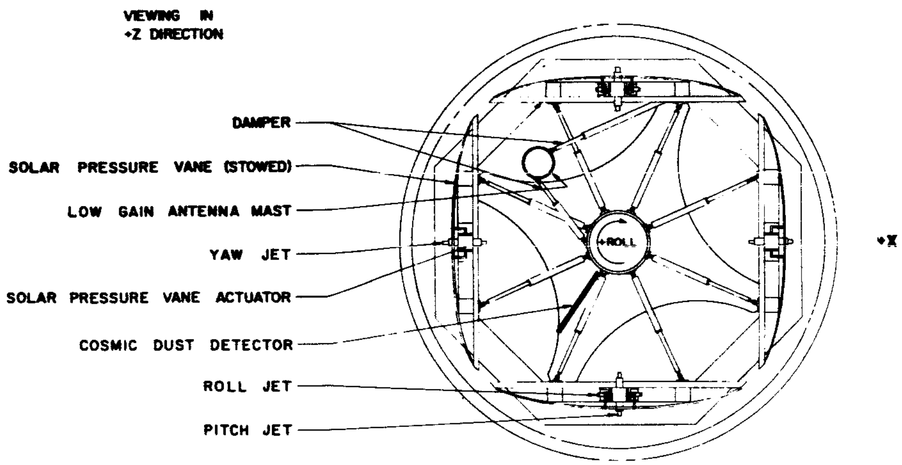
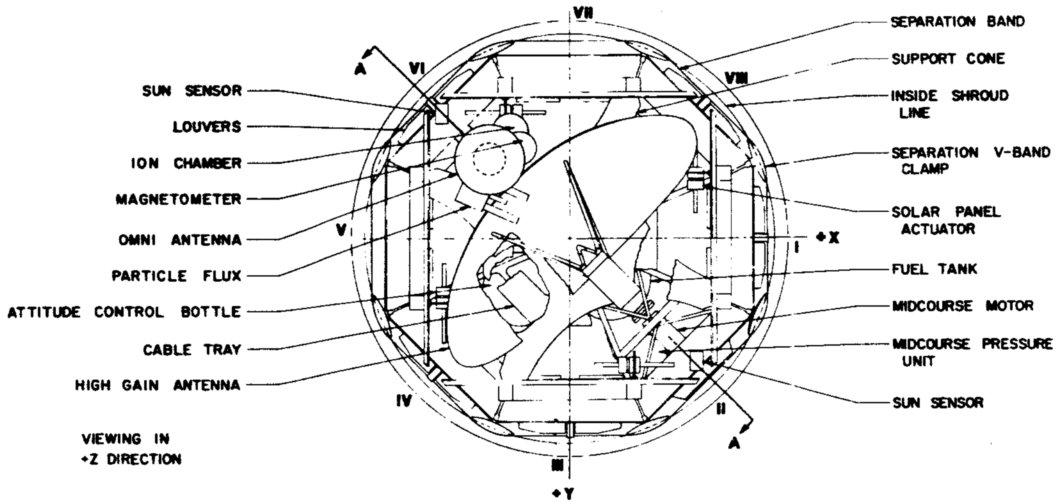
NOTE: IN THIS VIEW ONLY, SOLAR PANEL & LOW GAIN ANTENNA DAMPERS, A/C GAS JETS, SOLAR PRESSURE VANES, & SOLAR PRESSURE VANE ACTUATORS.

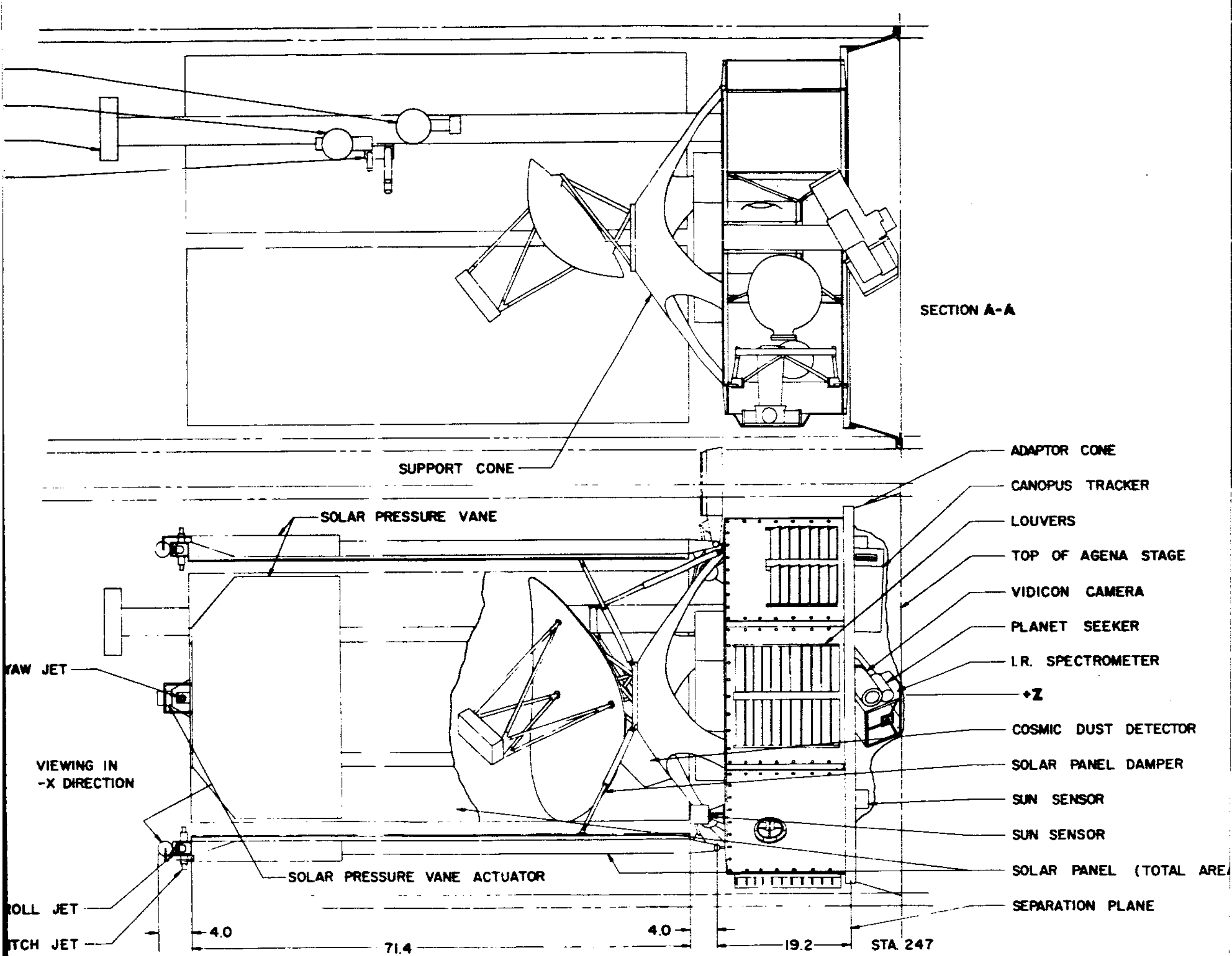


ION CHAMBER
MAGNETOMETER
LOW GAIN ANTENNA
PARTICLE FLUX

NOTE: SOLAR PANEL AND LOW GAIN ANTENNA DAMPERS, A/C GAS JETS, SOLAR PRESSURE VANE & SOLAR PRESSURE VANE ACTUATOR OMITTED IN THIS VIEW FOR CLARITY.

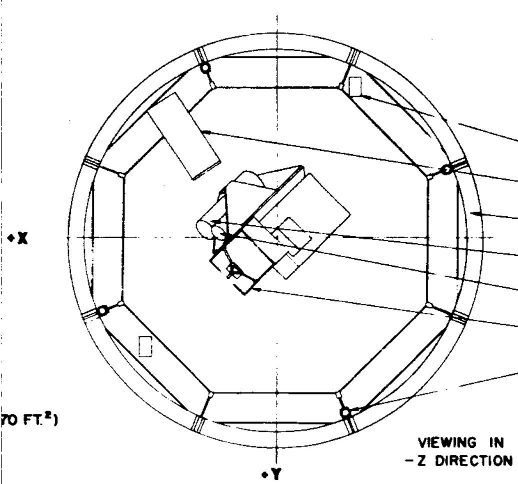


1


SECTION A-A
SUPPORT CONE
SOLAR PRESSURE VANE
ADAPTOR CONE
CANOPUS TRACKER
LOUVERS
TOP OF AGENA STAGE
VIDICON CAMERA
PLANET SEEKER
I.R. SPECTROMETER
+Z
COSMIC DUST DETECTOR
SOLAR PANEL DAMPER
SUN SENSOR
SUN SENSOR
SOLAR PANEL (TOTAL ARE
SEPARATION PLANE
AW JET
VIEWING IN
-X DIRECTION
SOLAR PRESSURE VANE ACTUATOR
OLL JET
TCH JET
4.0
71.4
4.0
19.2
STA 247

2

NOTES: UNLESS OTHERWISE SPECIFIED.
1. UPPER TEMPERATURE CONTROL HEAT SHIELD NOT SHOWN IN ANY VIEW.



(REF: JPL NO. 4190302)

Figure 3-1 General Arrangement of Spacecraft

3

E-3236-1

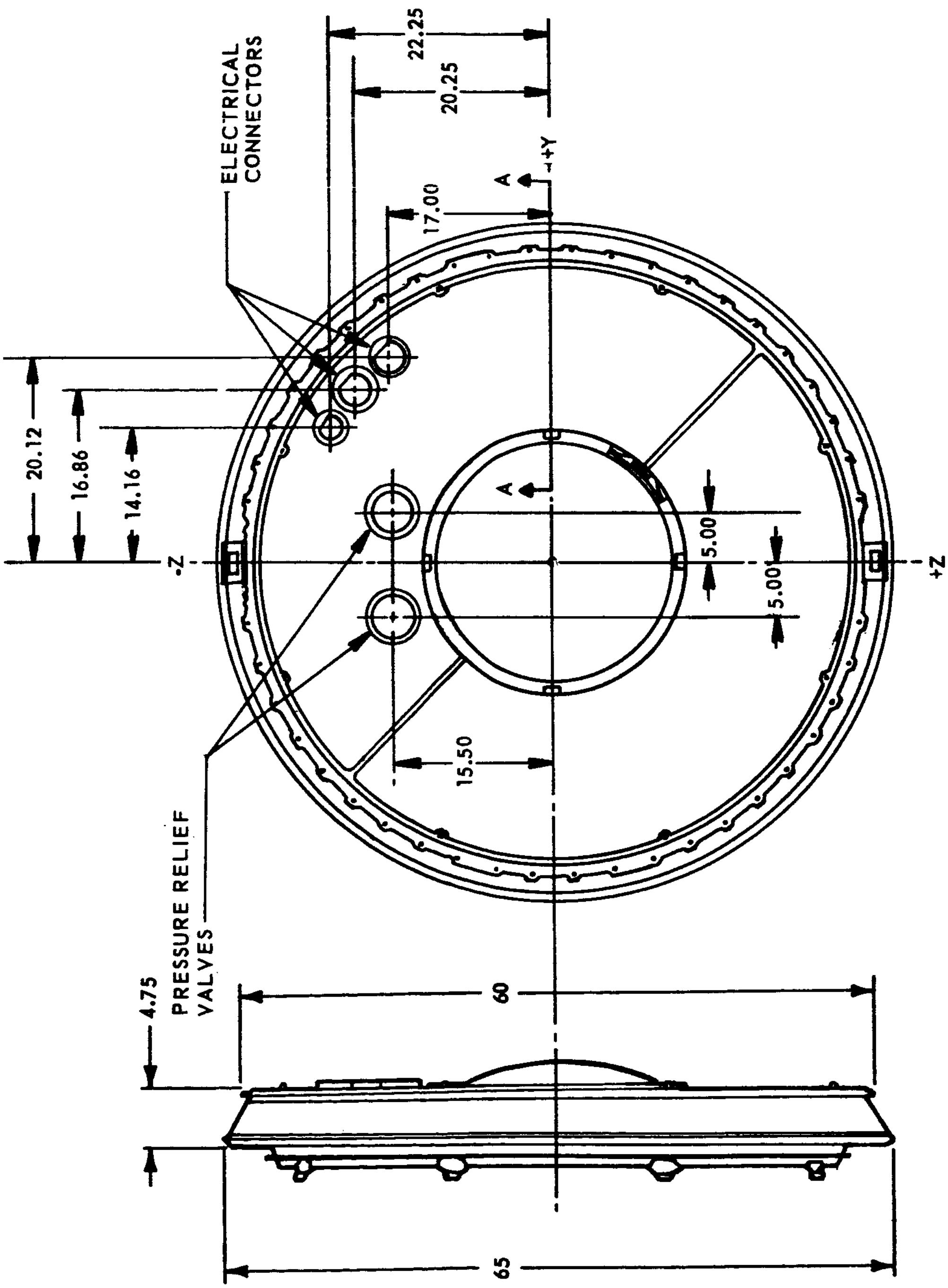
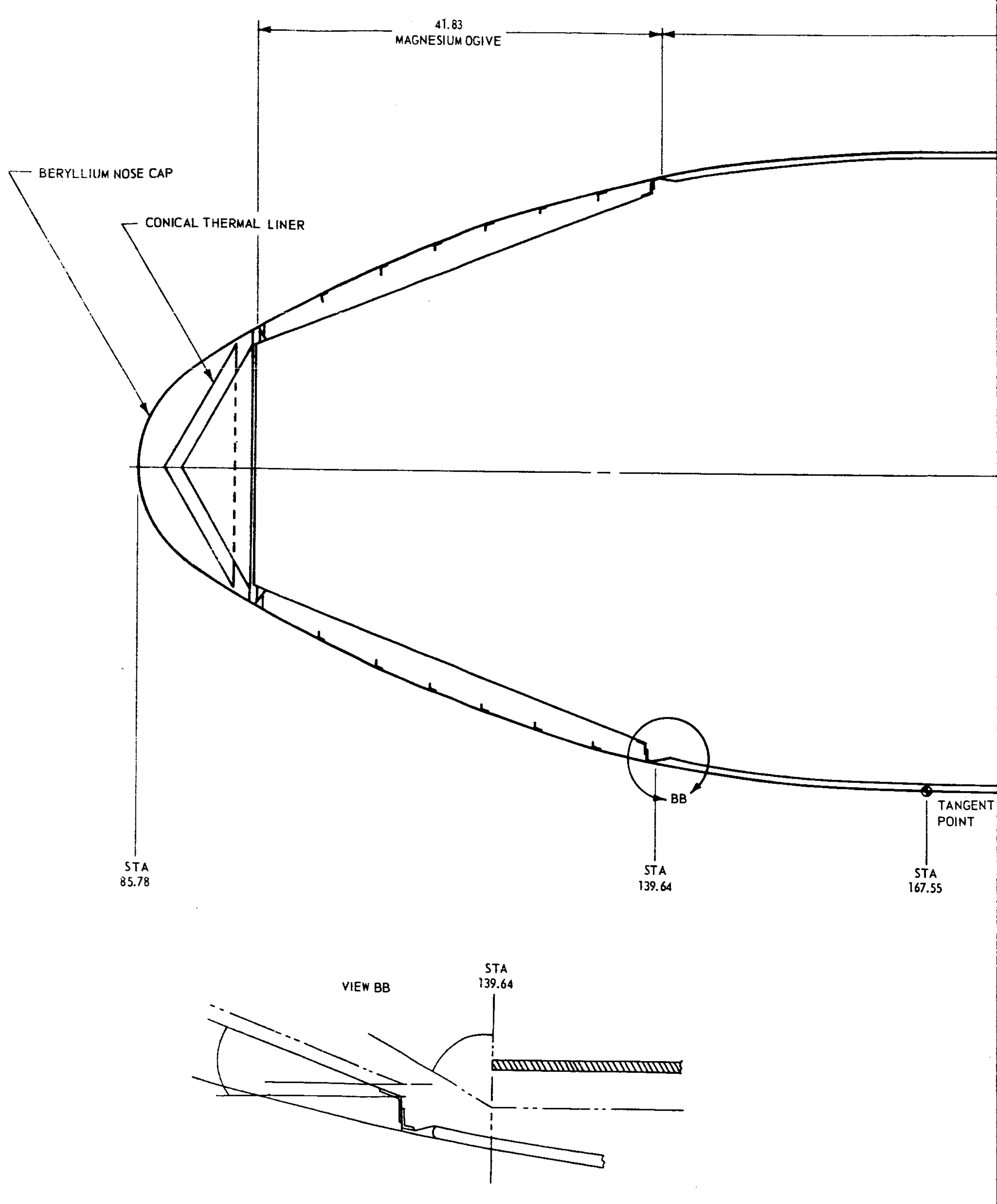


Figure 3-2 General Arrangement of Spacecraft Support Structure


41.83
MAGNESIUM OGIVE
BERYLLIUM NOSE CAP
CONICAL THERMAL LINER
BB
TANGENT
POINT
STA
85.78
STA
139.64
STA
167.55
VIEW BB
STA
139.64

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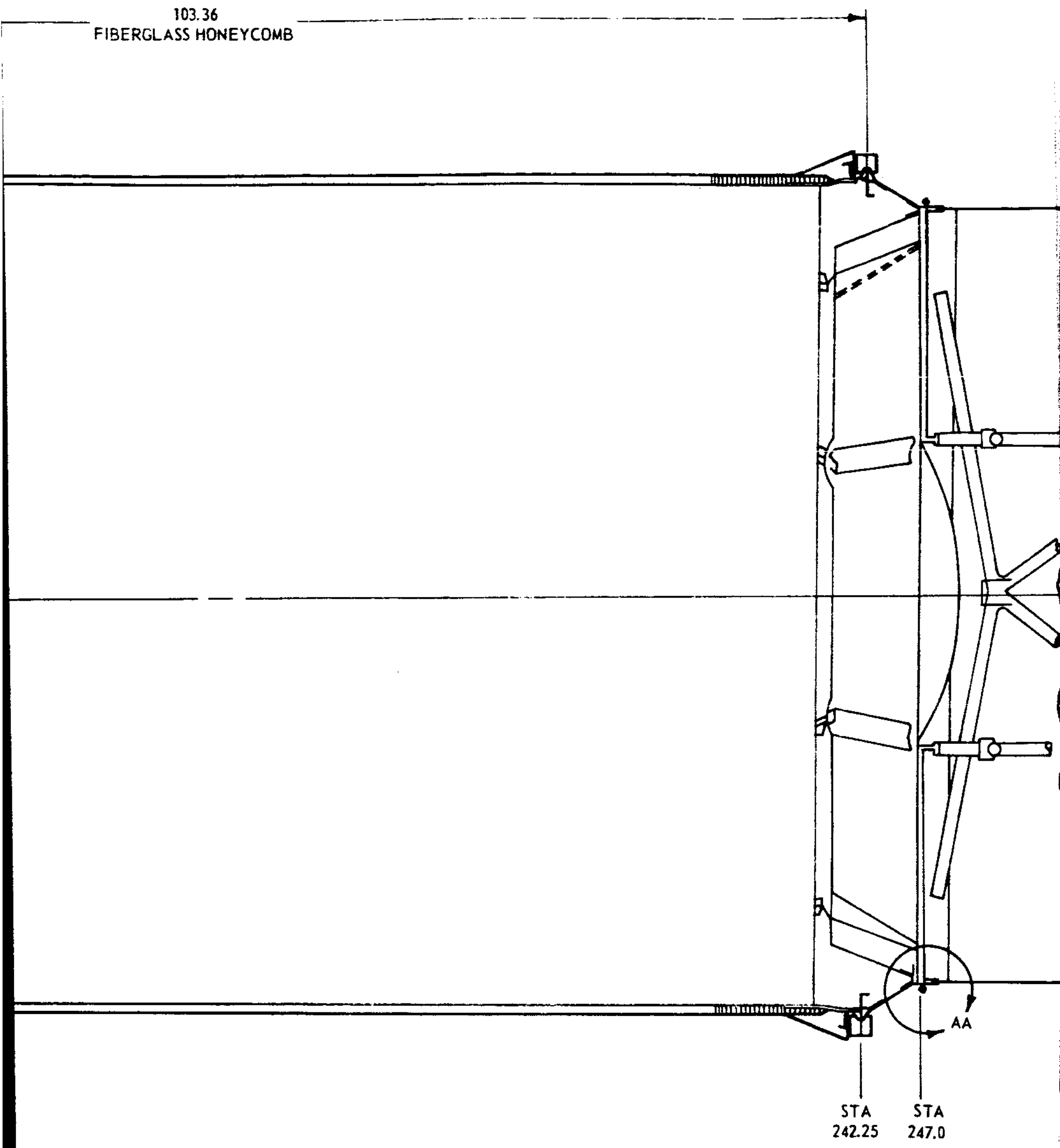

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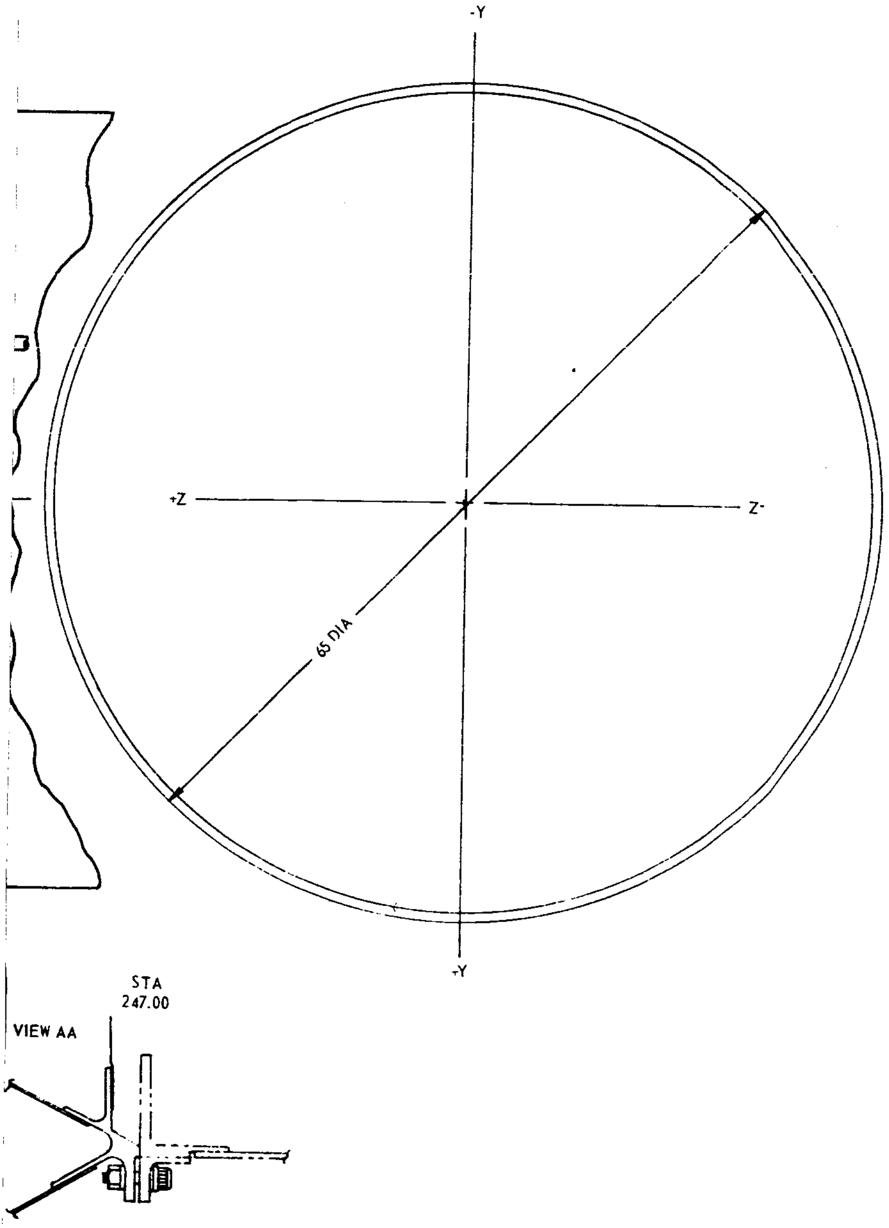

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Figure 3-3 General Arrangement of Aerodynamic Shroud

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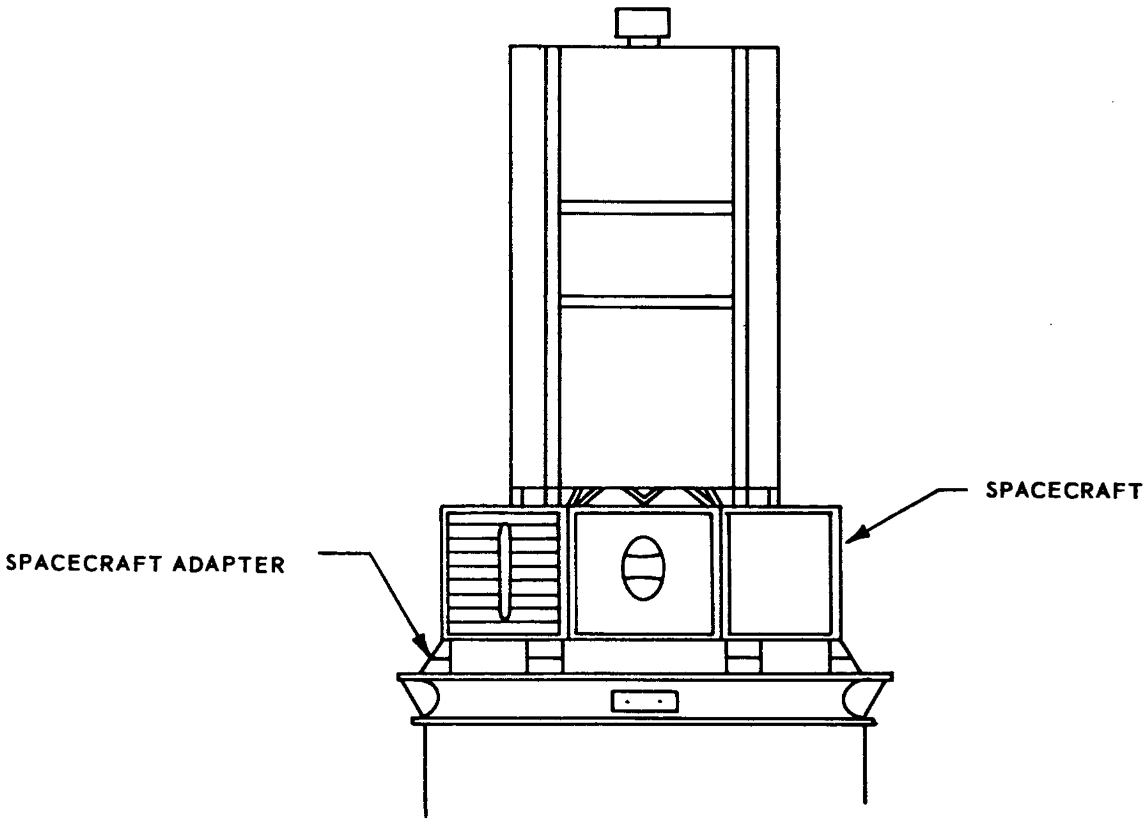

Figure 3-4 Composite Assembly of
Spacecraft and Shroud Transition Section

3.2 MASS AND STIFFNESS PROPERTIES

3.2.1 Mass Properties

Indicate the actual or target weights, CG locations, and moments and products of inertia for each of the listed major items and the tolerances thereon.

Mass distribution diagram: Provide Figure 3-5 to illustrate the spacecraft mass distribution versus length (station numbers).

3.2.2 Stiffness Properties

Provide Figure 3-6 to illustrate the spacecraft bending, shear, and torsional stiffness distribution factors, that is, the EI, AG, and JG products respectively.

3.3 MECHANICAL INTERFACE REQUIREMENTS

3.3.1 Alignment and Tolerance

Column (a): Define the alignment requirements by specifying the spacecraft axis (or axes) requiring alignment to the Agena axis, e.g., optical axis, maximum moment of inertia axis, etc.

Column (b): Identify corresponding Agena axis (axes) to which alignment is to be accomplished, e.g., spin table axis, horizon sensor vertical axis, etc.

Column (c): Give angle required between the two affected axes.

Column (d): Specify the tolerance on the alignment angle.

Column (e): Describe the method used to define the spacecraft axes, e.g., mirrors, alignment targets, etc.

3.3.2 Matchmate

A matchmate tool is defined as any structure designed and built specifically for the purpose of simulating a spacecraft to the extent necessary to demonstrate mechanical compatibility with adjoining structures. State all purposes for which a matchmate tool shall be employed.

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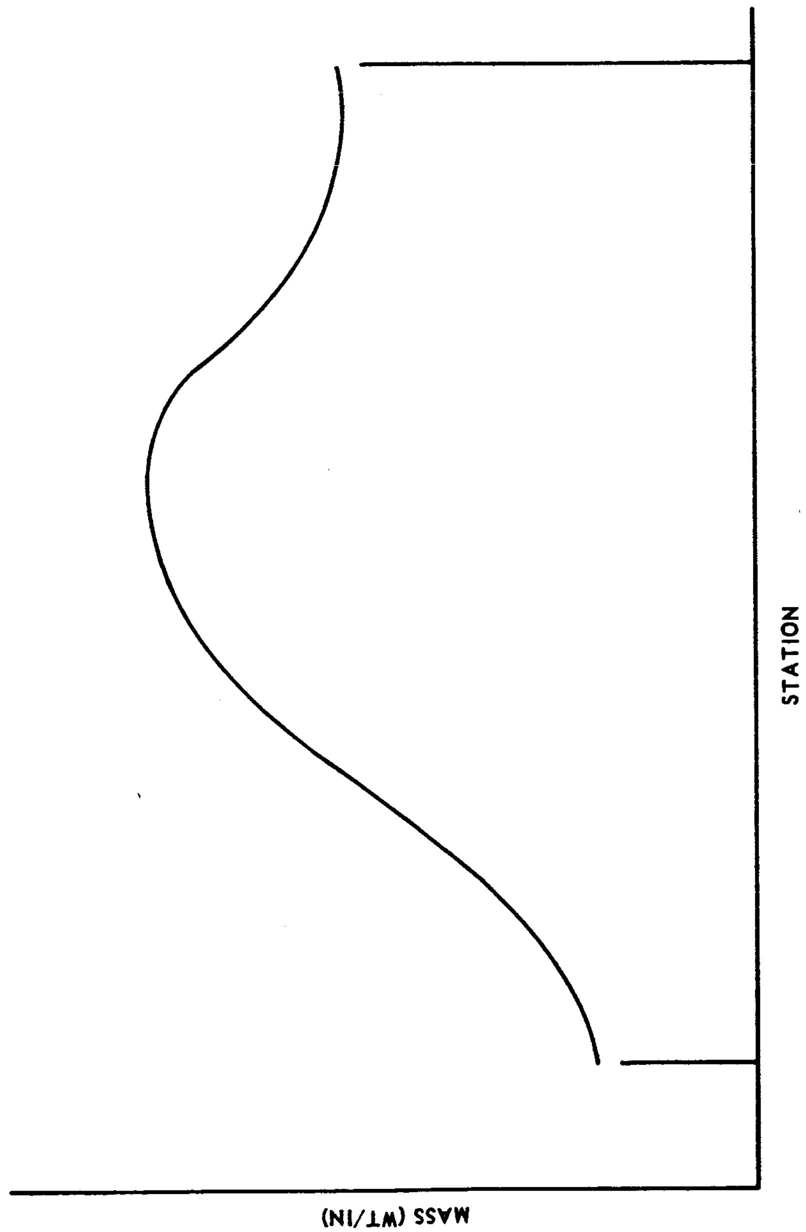

Figure 3-5 Spacecraft Mass Distribution Diagram

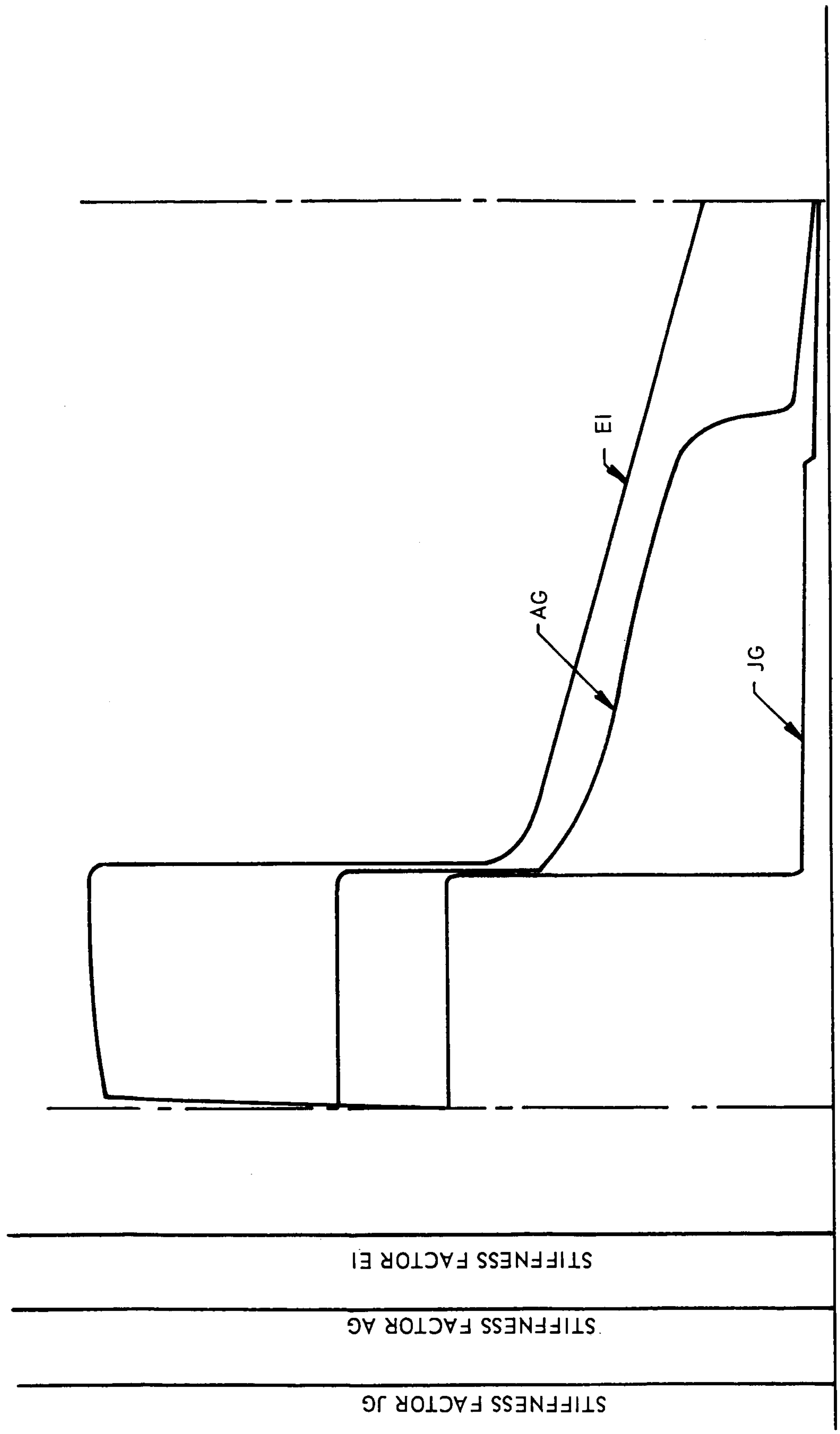

Figure 3-6 Spacecraft Structural Stiffness Factors

3.4 ELECTRICAL INTERFACE REQUIREMENTS

The electrical interface between the spacecraft and Agena requires the definition of the interface between the Agena and spacecraft support structure plus that of the interface between the spacecraft and spacecraft support structure. Standard hardware defined in Ref. 1 should be used. Usage of non-standard hardware to satisfy unique requirements must be justified.

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3.4.1 Spacecraft/Spacecraft Support Structure

Item (a): Specify the number of electrical interface connectors and connections required.
Item (b): Indicate method of electrical bonding of the mechanical interface, if required.
Item (c): Describe method used for electrical disconnect at spacecraft separation, e.g., guillotine and pull away plugs.
Item (d): Indicate any special considerations, e.g., bearing pads for separation switches, etc.

3.4.2 Spacecraft Support Structure/Agena

Complete the form by providing the necessary information as described under 3.4.1.

3.5 ELECTRICAL REQUIREMENTS

3.5.1 Instrumentation

Those items of flight and/or preflight test instrumentation which are mission peculiar requirements shall be specified. The listing shall include the instrumentation within the spacecraft which is to be telemetered over the Agena link (within the IRIG channel F capabilities) and the instrumentation installed in the Agena which is required by the spacecraft or mission, e.g. accelerometers. Agena capabilities are given in Ref. 1.

Column (a): Indicate the type of measurement, e.g., temperature sensor, acceleration, acoustic level, etc.

Column (b): For continuous channel measurements, specify the frequency response required. For commutated measurements, give the required minimum sampling rate.

Column (c): Give total sensing system accuracy required, i.e., from sensed parameter or transducer voltage to ground readout.

Column (d): Indicate range of measurement in terms of sensed parameter or transducer voltage.

Column (e): Specify flight phase during which monitoring of each parameter is required. Example: Vibration during Booster burn and Agena first and second burns.

Column (f): Describe the purpose of the measurement, e.g., temperature sensor to determine total heat flux through the shroud during injection.

Columns (g and h): Classify the measurements as mandatory or desirable. Indicate under "Desirable" the relative priority for each measurement by assigning the most important item a value of 1 and the others increasing numbers in order of priority.

Column (i): Specify location of the measurement by vehicle station, quadrant, and other detail as required. Alternatively, provide drawing.

3.5.2 Spacecraft Power to be Supplied by Launch Vehicle

In this section, specify the spacecraft power that is drawn from the vehicle power supplies. See Ref. 1 for definition of standard and optional power supplies.

Column (a): Indicate the spacecraft function that requires the power specified. Identify all squib-operated functions and any other circuit peculiarities.

Column (b): Give the load in watts and duty cycle in percent operating time drawn from the 28-volt unregulated battery bus.

Columns (c) thru (f): Supply information as described above and in addition give the power factor for the ac loads.

Column (g): For non-standard Agena power required by the spacecraft, give the type of power by specifying ac or dc. Specify the values for each parameter listed.

3.5.3 Switch Loading for Spacecraft Functions Activated Via Launch Vehicle

If the Agena provides programming of events in the spacecraft, the nature of the loads being switched is required under the heading of switch loading. The switch closure (or opening) provided can complete (or interrupt) power from the Agena or spacecraft power supply. (See Ref. 1.)

Column (a): Specify event accomplished by the issuance of the Agena command.
Column (b): Indicate whether normally open or normally closed circuit is required. In general, momentary open and close are not available without special design effort. Once activated, the switch will remain in the activated position throughout the remainder of the mission.
Column (c): Specify nature of the load; if squib, indicate whether it is of the fail-open or fail-short type and whether fusistors are included in the circuit.
Column (d): Give impedance of the circuit.
Column (e): Give voltage level to be switched.
Column (f): Indicate current drain, both peak and steady state.

3.6 ENVIRONMENTAL REQUIREMENTS

3.6.1 Spacecraft Thermal Environment

Specify the requirements of the spacecraft during ground checkout and flight to establish suitable spacecraft environment and to shape the ascent trajectories.

Items A1 and 2: Indicate the spacecraft bulk temperature limits during ground checkout and transportation. If bulk temperature limits are not an adequate specification to define the thermal environment, use other criteria and delineate critical items.

Item A3: Give the power dissipation in watts for the spacecraft during pad testing periods. Also give the cycle time, the number of cycles, and the frequency of cycles. Provide Figure 3-7 to indicate the spacecraft power dissipation during the testing period. (See sample illustration.)

Item B: Indicate the maximum permissible heat flux in Btu/hr ft^2 radiating from the inner surface of the shroud to the spacecraft during ascent.

Item C: Indicate any other thermal requirements or restraints.

Item D: Complete the table by complying with the following instructions.

Item (a): List those temperature sensitive spacecraft components which impose a restriction on launch time, ascent trajectory, or shroud jettison time.

Item (b): Give location by station and quadrant or reference to Figure 3-1.

Item (c): Give high and low temperature limits.

Item (d): Indicate average and peak power dissipation for each piece of equipment.

Item (e): Give the operating intervals with regard to liftoff or other significant mission event for each piece of equipment.

Items (f) and (g): Provide surface absorptivity and emissivity for those isolated pieces of spacecraft equipment subject to radiation from the shroud.

Item (h): Specify the heat capacity of equipment listed under items (f) and (g).

Item (i): Indicate basis for all requirements.

3.6.2 On-Pad Spacecraft Cooling

List any restrictions imposed upon the cooling media, e.g., temperature limits, type of gas, etc. If components of spacecraft are susceptible to damage through rapid pressure changes or excessive pressure differential, list components and pressure limitations. Specify the maximum specific humidity of the spacecraft cooling medium.

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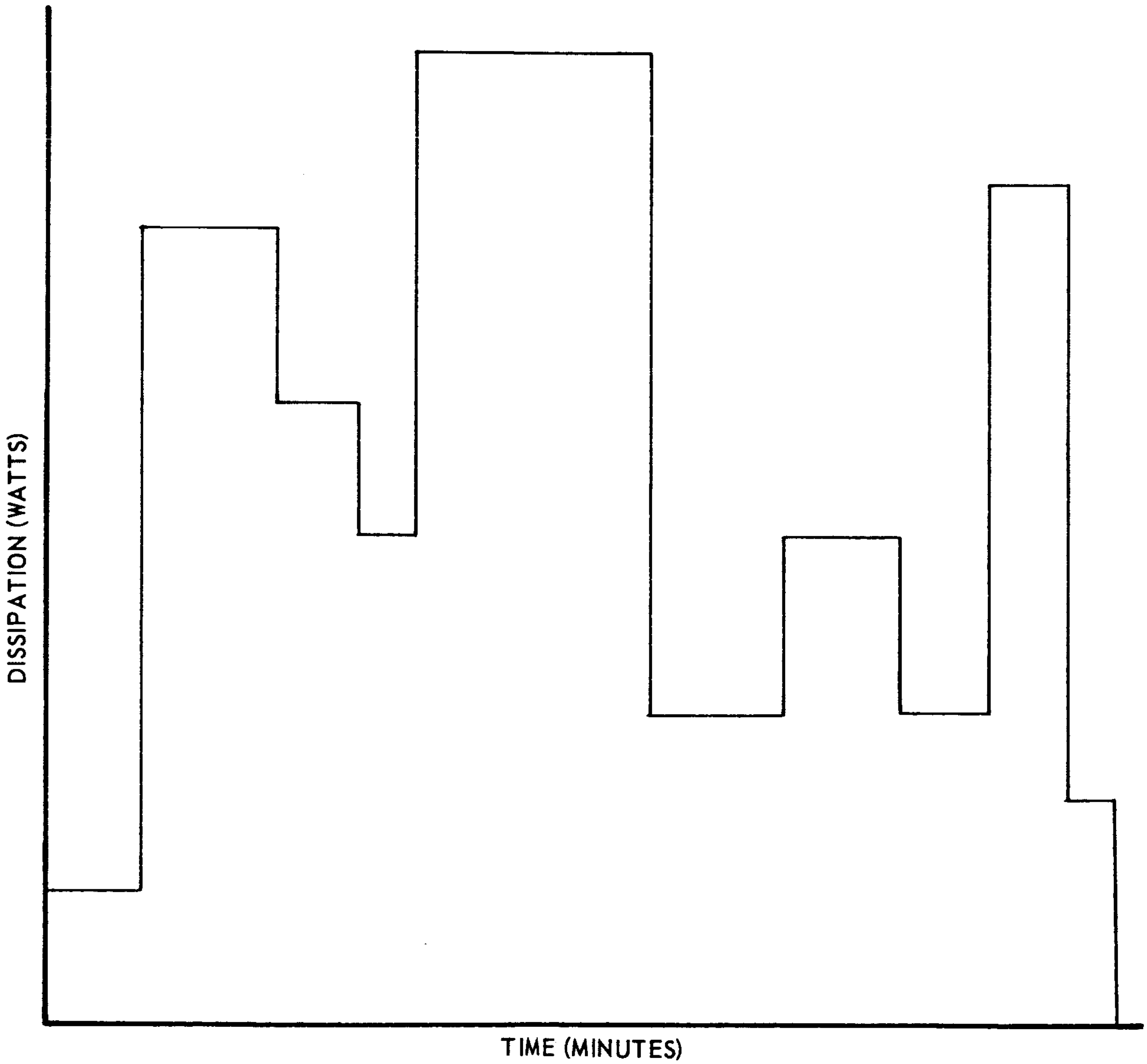

Figure 3-7 Spacecraft Power Dissipation During Pad Testing Periods

E-3236-1

3.6.3 Contamination Control

A. Optical Degradation

Complete the indicated blanks to specify the optical transmission degradation allowed to occur from accumulation of contaminants from ground cooling gases, pyrotechnic operations, or materials outgassing.

B. Water Vapor

Give the allowable specific humidity of the air conditioning and purging gases.

C. Contaminants

Specify the allowable concentration of contaminants by completing the table.

Item (a): Specify the critical locations in terms of vehicle coordinates.
Item (b): Identify the permitted ranges of particle sizes in the gases introduced into the spacecraft area. Suggested ranges are 25-50, 50-100, 100-150, greater than 150 microns. Indicate specification for both metallic and nonmetallic contaminants.
Item (c): Give allowable concentration for each particle size range. Concentration may be measured as particles per square inch of a standard filter for a standard volume of gas flow.
Item (d): Identify an acceptable method to be used in determining permitted contaminant concentration in (c).
Item e: Give the basis of requirement, tolerance, or limitation.

3.6.4 Sterilization

Indicate the requirement imposed on the launch vehicle by the spacecraft sterilization requirements.

Item (a): Indicate those areas of the launch vehicle involved.
Item (b): Specify the type of sterilizing medium or technique to be employed.

Item (c): Give the time required to accomplish the sterilization process and the time before launch when initiation of the process is required.

Item (d): Specify method of introduction of the sterilizing medium into the area to be treated.

3.6.5 Applied Loads

List any spacecraft acceleration limitations. See Ref. 1 for launch configuration specifications.

3.6.6 Acoustic Noise

Specify any acoustic noise limitations or requirements imposed by the spacecraft if less than those listed in Ref. 1.

3.6.7 Electromagnetic Environment

The specification of the electromagnetic environment contributed by the spacecraft and the susceptibility of the spacecraft equipment to electromagnetic interference is necessary for establishing preliminary interference criteria needed to assure trouble-free operation of the combined launch vehicle. Associated factors requiring definition are the grounding, shielding, and bonding techniques used in the design of the spacecraft, the magnetic materials requirements placed on the Agena structure, and the nuclear radiation environment.

A. Conducted Interference

Column (a): Indicate the interface conductor by connector designation (Section 3.5) and pin letter if possible.

Column (b): Specify fundamental noise frequency or the time duration and repetition rate of the transients.

Column (c): Indicate the RMS voltage level of the noise component on the interface conductor.

Column (d): Indicate the peak voltage of the noise signal component or transient.

Column (e): List the source impedance for the circuit on the spacecraft side of the interface.

B. EMI or RFI Tests

List any required spacecraft EMI or RFI tests involving the launch vehicle. A policy statement in regard to EMI and RFI testing is given in Ref. 1.

C. Radiated Interference

Column (a): Identify all radiating equipment (other than transmitters) within the spacecraft.

Column (b): Give location of each equipment by vehicle station and quadrant or refer to Figure 3-1.

Columns (c) and (d): Provide radiated voltage levels and corresponding frequency or frequency band as measured in accordance with MIL-I-26600.

D. Spacecraft Transmitter Identification (For transmitter operating prior to injection)

Column (a): List make and model of each spacecraft transmitter.

Column (b): Specify rated transmitter power output.

Column (c): Indicate measured transmitter power output.

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Column (d): Specify type of antenna, e.g., slot, dipole, horn, etc.
Column (e): Indicate location of antenna by vehicle station and quadrant or supply information on Figure 3-1.
Column (f): Give range of operating frequency.
Column (g): Specify the transmitter bandwidth.
Column (h): Specify operating periods of each transmitter in terms of time from liftoff, or other significant mission event.
Column (i): Indicate type of modulation used e.g., FM/FM, PAM, etc.
Column (j): Indicate method used to multiply the basic oscillator frequency to achieve the transmitter frequency. Provide multiplication ratio.

E. Transmitter Frequency Spectra

Complete the columns in accordance with applicable standards (see Ref. 3, 4, and 5).

Column (a): Identify each transmitter.
Columns (b) and (c): Provide abbreviated tabular frequency spectra data for each transmitter in terms of db below the fundamental output. The frequency band shall extend from 150 kc to 10 kMc.

F. Spacecraft Wiring Design

Item (1): Define the electrical grounding system employed in the spacecraft design and the necessity for connection to the Agena single point ground.
Item (2): Specify extent to which shielded wire is used in the spacecraft circuits for power, signal, and pyrotechnics. State any exceptions to the generalized design intent.
Item (3): Describe technique used for handling spacecraft signal, power, and pyrotechnic ground returns, e.g., parallel or twisted runs used, grounds returned through the structure, etc. Specify means for handling wire shield grounds and equipment case grounds.
Item (4): Indicate methods of electrical bonding used throughout the spacecraft.

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Item (5): Describe design philosophy with respect to grouping similar voltage level circuit wires. Consider low and high level signal circuits, power, and pyrotechnics.

G. Magnetic Materials

Item (a): Specify whether a requirement exists for limiting use of ferromagnetic material in the Agena structure.
Item (b): If required, specify maximum allowable permeability and indicate location of material in terms of spacecraft coordinates.

H. Nuclear Radiation

Column (a): Indicate type of nuclear radiation emitted by the spacecraft, e.g., neutron, proton, α, β, γ.
Column (b): Indicate energy distribution of the radiation in million electron volts (MEV).
Column (c): Specify dose rate in roentgen/unit time or particles/cm^2/unit time.
Column (d): Give total integrated dose rate.
Column (e): Give integration time for dosage specified in Column (d).
Column (f): Provide identification of vehicle location (station number) where the specified dose and dose rate apply.
Columns (g) and (h): Indicate continuous or intermittent nature of radiation.
Column (i): Give duration of continuous operation and starting time referred to liftoff, or the duration of the intermittent operations and their repetition rates.

3.6.8 Miscellaneous Environmental Considerations

List all environmental requirements and restraints other than those specified in paragraphs 3.6.1 through 3.6.6, for example, vibration levels, pressure differentials, rate of pressure release, etc.

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3.7 CLEARANCE REQUIREMENTS

3.7.1 Static Clearances

Specify all static clearance requirements between the spacecraft, aerodynamic shroud, and spacecraft support structure. Where clearance requirements are complex, a figure should be included if it provides greater clarification.

3.7.2 Dynamic Clearances

Specify all dynamic clearance requirements between the spacecraft, aerodynamic shroud, and spacecraft support structure envelopes. Since clearance requirements are complex, a figure should be included. Figure 3-8 is provided as an example. (Total deflections include flight bending, vibration, and production tolerances.)

3.8 SEPARATION REQUIREMENTS AND RESTRAINTS

The separation requirements and restraints under this heading are those for shroud separation and spacecraft separation.

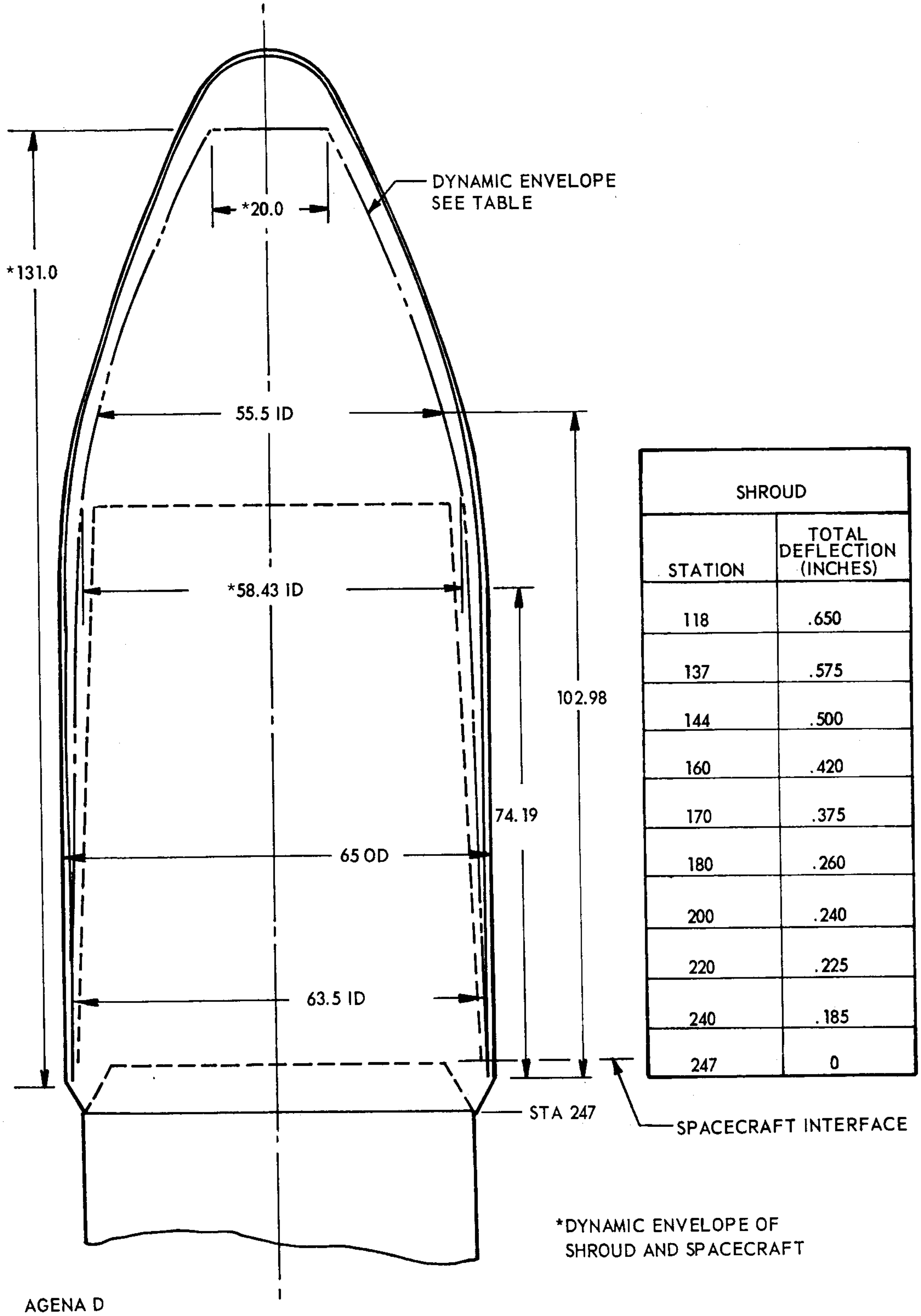
3.8.1 Shroud

Indicate peculiar requirements or restraints. Identify and define any required characteristics that are not provided by the standard components available in Ref. 1.

3.8.2 Spacecraft

Vehicle attitude rates are to be specified in terms of rotation about the corresponding body axes. The angles are defined by Figure 3-9 and the following:

> Yaw is the angle measured in the local horizon between the projection of the vehicle longitudinal axis and the orbit plane.



SHROUD	
STATION	TOTAL DEFLECTION (INCHES)
118	.650
137	.575
144	.500
160	.420
170	.375
180	.260
200	.240
220	.225
240	.185
247	0

Figure 3-8 Aerodynamic Shroud Dynamic Envelope

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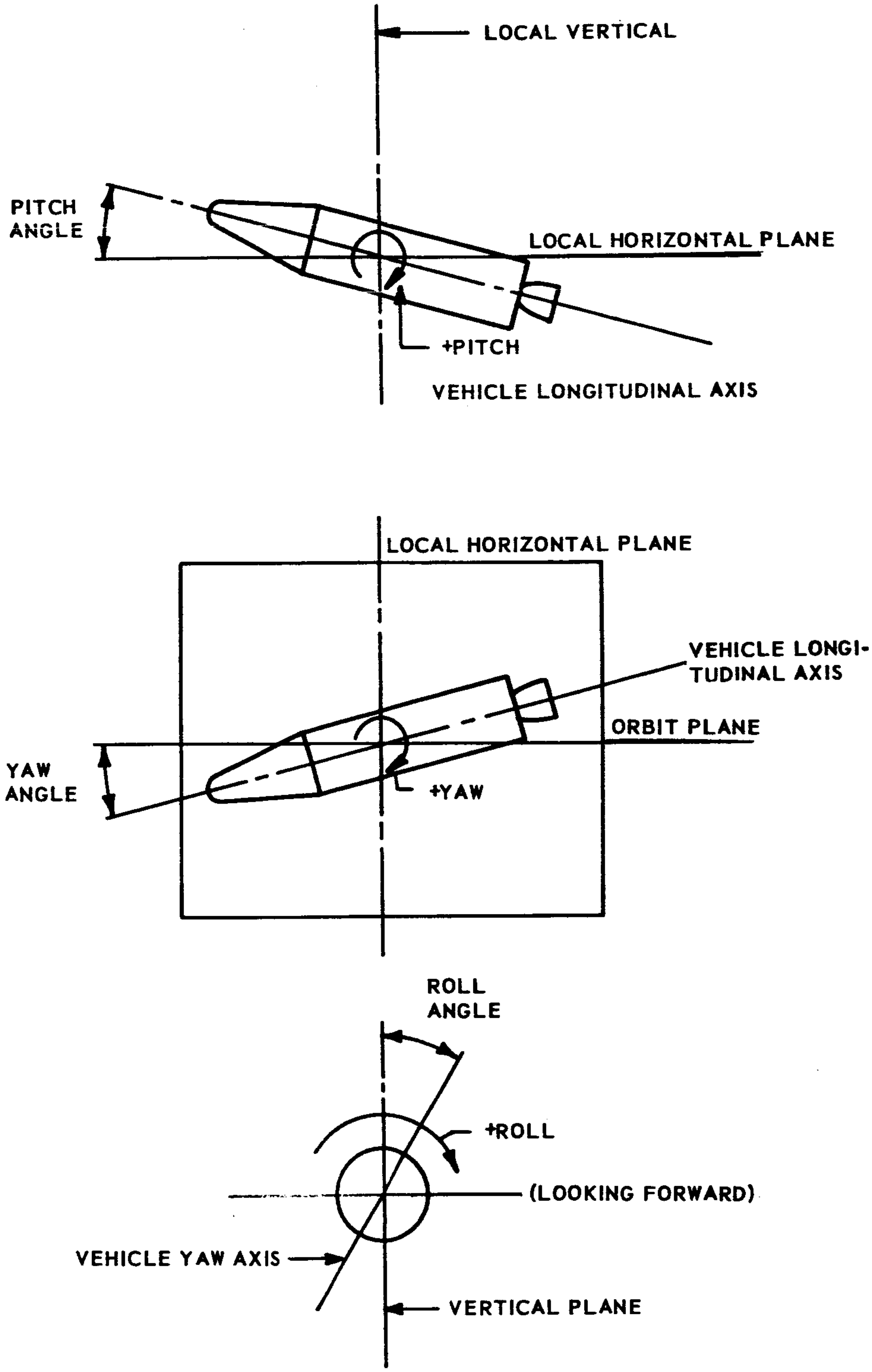

Figure 3-9 Definition of Vehicle Angles

E-3236-1

Pitch is the angle measured in the local vertical plane between the vehicle longitudinal axis and the local horizontal plane.

Roll is the angle measured between the local vertical plane and the yaw axis in a plane normal to the vehicle longitudinal axis.

a. At spacecraft separation indicate the angle and angular rate requirements on vehicles.

b. If the spacecraft is to be spin stabilized, indicate the allowable wobble angle in degrees (precession half cone angle--the angle between the body fixed spin axis and the angular momentum vector).

c. Indicate the required spin rate (angular velocity about the spin axis) in rad/sec. Also indicate tolerances.

d. Indicate the angular acceleration limit during spin-up in rad/sec.

e. In remarks column, note any special requirements such as separation velocity, acceleration, and limits desired.

NOTE:

Figures may be used for clarification.

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SECTION 4

PROGRAM-PECULIAR TEST REQUIREMENTS

4.1 TEST REQUIREMENTS

Program-peculiar requirements may necessitate testing by the organization responsible to LeRC to demonstrate a capability required by the spacecraft. Also, tests may be required upon integrated specimens composed of the spacecraft and produced/supplied hardware from organizations responsible to LeRC to verify physical mating and operating characteristics. For each test required specify:

a. The type of test
b. The test objectives
c. The major components in the test assembly, e.g., spacecraft, spacecraft support structure, Agena D forward equipment rack
d. Any special procedures or test techniques that must be employed.

NOTE:

Supply a paragraph number for each test listed, i.e., 4.1.1, 4.1.2, etc.

4.2 STUDY REQUIREMENTS

Program-peculiar requirements may necessitate special studies by the organization responsible to LeRC. For each study required, specify:

a. Type of study
b. The study objectives
c. Any special techniques of method or analysis that should be employed
d. Reporting requirements

NOTE:

Supply a paragraph number for each study listed, i.e., 4.2.1, 4.2.2, etc.

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SECTION 5
LAUNCH BASE REQUIREMENTS AND RESTRAINTS

5.1 TRANSPORTATION AND HANDLING CRITERIA

The transportation and handling criteria specified herein shall pertain to those items which are to be handled or transported by organizations responsible to LeRC but which are not LeRC's responsibility to have manufactured. It is assumed that no transfer of restraints is necessary if the manufacturer is also responsible for all handling of his items. However, this section includes operations such as hoisting into position for stack-up assembly of the launch vehicle system on the pad. This operation is usually defined as the entire responsibility of one contractor. See Ref. 6, 7, 8, 9, and 10.

Column (a): Indicate those items which require transport and handling by organizations responsible to LeRC, e.g., spacecraft, spacecraft assemblies, etc.
Column (b): Specify type of power required during transport, e.g., 28v dc, 115v 400 cps. Indicate areas between which transportation is to occur, e.g., MAB to pad.
Column (c): Specify the environmental control required during transport. For air conditioning, specify air temperature, flow rate, humidity, and filtration requirements.
Column (d): Specify other requirements, e.g., shock limitations, orientations,etc.

5.2 UMBILICAL AND TEST PLUGS

5.2.1 Electrical Umbilical

Column (a): Assign an item number to each function requiring an umbilical conducter. Individual ground leads are considered unique functions requiring an item number.

Column (b): Give a descriptive title to each function.

Column (c): Indicate whether the monitored function is continuous. If the function is not continuous, specify the duty cycle.

Column (d): Indicate nature of shielding required for the umbilical conductor.

Column (e): Give the range of signal or power voltage to be carried.

Column (f): Specify current level

Column (g): Indicate signal or power frequency or frequency range.

Column (h): Give spacecraft load impedance,including both real and imaginary components.

Column (i): Specify nature of load, e.g., resistance bridge, motor winding, squib, etc.

Column (j): Indicate launch pad monitoring (or supply) area, e.g., Launch Operations Building (LOB), Blockhouse (BH), Launch Pad Building (LPB), Pad Electrical Building (PEB), etc.

Column (k): List type of monitoring required at the termination, e.g., meter, oscillograph, etc.

Column (l): Classify the nature of the umbilical requirement as either mandatory or desirable. For all items indicated to be desirable, enter their relative priority, starting with 1 (highest priority) and continuing to the lowest priority, equal to the number of functions in this classification.

5.2.2 Electrical Test Plugs

The electrical test plugs differ from the umbilical in that they are removed from the launch vehicle at the time of gantry roll-back (approximately launch minus two hours). The umbilical remains attached until committment to flight. Since umbilical conductors are severely limited in number, the test plugs should be used for all functions that do not require last minute monitoring or control. Complete the table as indicated under paragraph 5.2.1.

5.2.3 Pressurized Gas Loading Umbilicals

Complete the columns in the table as follows:

Column (a): Indicate type of gas to be loaded, e.g., nitrogen, helium, ammonia, etc.

Column (b): Give specification requirements, including solid contaminants, purity, humidity, etc.
Column (c): Specify the maximum temperature and pressure allowed in the prelaunch periods. Include considerations of personnel safety.
Column (d): Specify flight load limits of temperature and pressure.
Column (e): Give total volume of gas to be loaded, including contingency for possible leakage, flight holds, etc.
Column (f): Specify gas dumping and venting requirements.

5.2.4 Propellant Loading Umbilicals

Provide propellant data by completing the columns in the table (also see Ref 11).

Column (a): Give nature of propellant together with its specification on purity, filtering, etc.
Column (b): Specify loading flow rates required. If different rates are required for initial load and topping, so specify.
Column (c): Indicate loading pressure or pressure range required.
Column (d): Give temperature limits for loading of propellants and for loaded propellants.
Column (e): Give total mass required, including contingency for boil off, launch holds, etc.
Column (f): Specify dumping and venting requirements.

5.2.5 Parasitic Coupler and Reradiating Antennas

List each antenna and coupler; give locations, allowable attenuation, stability, and repeatability.

5.3 LAUNCH BASE SEQUENCING

In order to allow test planning and prepare for the handling of the launch vehicle at the launch base, an outline of the prelaunch program-peculiar activities is required. A typical launch vehicle sequence of activities is given in the LMSC Catalog (Ref. 1).

5.3.1 Spacecraft Assembly Building Operations

Column (a): Specify, in chronological order, spacecraft tests or operations to be performed.

Column (b): Specify duration and placement of time period for operations with regard to launch date.

Column (c): Specify support equipment required and supplier of equipment, i.e., spacecraft-contractor supplied, or supplied by organizations responsible to LeRC.

Column (d): Description of expected support, i.e., handling, transportation, alignment, cooling, etc.

Column (e): Specify restraints imposed by operations on other launch base activities, and on validation times for spacecraft equipment.

5.3.2 Pad Checkout (Gantry in Place)

Column (a): Specify, in chronological order, spacecraft tests or operations to be performed.

Column (b): Specify duration and placement of time period required for operations with regard to launch date.

Column (c): Specify support equipment required and supplier of equipment.

Column (d): Describe the expected support, including air-conditioning requirements, spacecraft mating and de-mating requirements, etc.

Column (e): Specify communication requirements, including support equipment, i.e., parasitic antenna, antenna coupler, coaxial switch, etc.; and communication schedules, i.e., launch vehicle telemetry schedule and spacecraft RF transmission schedule.

5.3.3 Countdown Activity

Provide the sequence of operations required for the spacecraft in the final countdown period. Consider the following events:

Item (a): Periods of RF transmission by the spacecraft
Item (b): Spacecraft fueling and topping
Item (c): Switch from ground power to internal power
Item (d): Others

5.3.4 Pad Cabling Requirements

Indicate the pad cabling requirements by designating the number of conductors, wire sizes, and shielding needed between:

Item (a): The spacecraft and the launch pad building
Item (b): The spacecraft and the launch operations building
Item (c): Launch pad to launch operations buildings
Item (d): Other cable routings.

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REFERENCES

1. LMSC-A604166, Agena D Mission Capabilities and Restraints Catalog
2. AFMTCR 80-7, Airborne Flight Termination Systems
3. MIL-STD-442A, Telemetry Standards for Missiles and Aircraft
4. IRIG Document No. 106-60, Telemetry Standards
5. LMSC-447969, EMI Control Requirements and Electrical Interface for Agena Systems
6. LMSC-920493, Interface Control Requirements Specification for Space Systems Aerospace Ground Equipment
7. LMSD-1410595, General Environment Specification for Satellite Systems, AGE
8. USAF MIL-A-8421B, Air Transportability Requirements
9. USAF MIL-M-008090D, Mobility Requirements, GSE
10. ICC, Motor Carrier Safety Requirement.
11. AFMTCR 80-2, General Range Safety Plan, Volume I, "Pre-launch Safety Procedures"

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REVISIONS AND UPDATING

This document will be revised as necessary to maintain it in an accurate, up-to-date status. When such revisions are made, revised or addendum pages will be provided to holders of this document. Each such revised or addendum page will be marked in the upper right hand corner with the date of issue, and each issue will be accompanied by a Report Change Record indicating affected pages and instructing the recipient. The Report Change Record format is shown on the following page.

August 24, 1965

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LERC AGENA PROJECT-

SPACECRAFT CENTER INTERFACE OPERATING PROCEDURES

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LeRC Agena Project-

Spacecraft Center Interface Operating Procedures

1.0 Purpose of Operating Procedures

These operating procedures were developed to:

1. Establish a standard set of procedures to prevent misunderstandings between government agencies.

2. Permit the execution of government business in an orderly and efficient manner.

3. Provide information to all organizations concerned with a mission.

2.0 Launch Vehicle System Requirements and Restraints Document (RRD)

2.1 Significance

The purpose of the RRD is to set forth those technical requirements and restraints imposed by the mission upon the launch vehicle, associated L/V AGE, designated launch complex, and range. The launch vehicle is composed of the booster, booster adapter, Agena vehicle, shroud system and any structure or equipment supplied by the L/V Contractor in support of the spacecraft.

The RRD has been established by the LeRC Agena Project to be a standardized method of recording the s/c requirements and restraints on the launch vehicle system in missions employing the Agena D (SS-01B) vehicle. An approved version of the RRD is incorporated in LeRC launch vehicle adaptation and integration contracts as a work effort compliance requirement.

2.2 Format

A standard format for the RRD is provided by the "Agena Missions Standard Requirements and Restraints Document, Vol. I - Format and Vol. II - Instructions." Copies of this document can be obtained by writing the Agena Project, Attention: Plans and Programs Office, LeRC. The detailed instructions for filling out the format are provided in Vol. II and the RRD is to be composed in conformance with the format. The type of information required is the technical aspects of the s/c and missions as they affect the launch vehicle and attendant equipment. The information should be expressed in terms of thermal limitations, power requirements, interface details, orbit requirements, etc., of the spacecraft. Schedules, management plans, vehicle test procedures, L/V AGE designs, etc., should not be injected into the RRD. Actual s/c requirements and restraints rather than vehicle capabilities or limitations should be specified. Deviations from the format should only be made to amplify or elaborate upon s/c requirements and restraints.

2.3 Responsibility

It is the responsibility of the Spacecraft Center to originate the RRD by inserting the requisite information and transmitting a preliminary version to LeRC. The Agena Project is responsible for furnishing information pertaining to the launch vehicle systems and for the performance of limited studies needed to generate or qualify such information. The Spacecraft Center is referred to the "NASA Agena D Mission Capabilities and Restraints Catalog, LMSC A604166" which contains a compilation of the launch vehicle performance capabilities and characteristics, available standard hardware, basic descriptions of equipment, mission activities, etc., for missions employing the Agena D (SS-01B) vehicle.

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2.4 Schedule

As the Agena launch vehicle adaptation cycle depends upon the type of mission, the date the first draft of the RRD is required by LeRC varies. For the mission that required multiple launches and many peculiarized items of equipment, the RRD should be received by LeRC 22 months prior to launch. For uncomplicated, simple missions where the major portion of the peculiar equipment is already designed and/or built (repeat missions or missions using largely standard equipment), the RRD should be received by LeRC 18 months prior to launch. When development of major, new hardware is required, additional time beyond the 22 months schedule is needed by LeRC.

2.5 Definition Meeting

Prior to generation of the first draft of the RRD by the S/C Center, the S/C Center is encouraged to request a meeting with LeRC to discuss the mission and particulars to be covered in the RRD. LeRC may support such a meeting, if appropriate, with contractor personnel in addition to LeRC personnel.

2.6 Review and Approval

The sequence of events to transact an approved RRD is nominally as follows:

1. Preliminary RRD received by LeRC (ten copies should be supplied).

2. Preliminary RRD reviewed by Agena Project Management and technical personnel assigned to mission.

3. Comments are compiled and transmitted in writing to S/C Center.

4. A meeting between Agena Project and S/C Center personnel is scheduled by mutual agreement to resolve areas of disagreement and transmit technical information.

5. S/C Center personnel prepare a revised version of the RRD and transmit

same to LeRC with an approval page for the Agena Project Manager and Mission Manager signatures.

6. S/C Center receives approval page and suggested revisions, if any, and distributes approved RRD.

7. The RRD requirements become binding between the Agena Project and the S/C Center once the document has been approved by the Agena Project Manager and the Mission Manager.

8. Copies of the RRD may be submitted to the L/V Contractor for review at the discretion of the Agena Project. The S/C Center should, therefore, be in a position to transmit 15 copies of the RRD to LeRC for this purpose.

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2.7 Revisions

A revision control sheet is issued with the revised page(s). The RRD format described in para. 2.2 contains a revision control page. Proposed revisions are transacted per the sequence outlined in para. 2.6. However, Steps 1 to 3 may be circumvented by discussing the proposed revision at an official interface meeting. The revision submitted in Step 5 should be in the exact language proposed for insertion in the RRD.

2.8 Supplemental Information

1. Reference material not available to all parties concerned should not be included in the RRD.

2. The "Interface Control Drawings" (discussed in Section 3 herein), although not an integral part of the RRD, do serve to supplement the information provided in the RRD. These drawings document the agreed upon detailed implementation of the RRD requirements.

3. The "Interface Plan and Schedule" (discussed in Section 4 herein) also serves to supplement the information provided in the RRD. The RRD should contain only a listing of the s/c interfacing events that are to be included in this plan but no schedules for the accomplishment of these events. Schedules and other details are developed during an official interface meeting early in the development cycle of the launch vehicle.

3.0 Interface Control Documentation

3.1 Definition

Interface Control Documentation is defined as that documentation which serves to describe in detail all the physical characteristics and functional requirements of interfaces among systems and major assemblies designed or provided by associate contractors, system contractor and/or government agencies. The documentation shall specify all critical characteristics of mechanical, electrical, hydraulic, pneumatic, optical, RF, type of material, and weight requirements at the interface. The documentation includes but may not be limited to mechanical and electrical interface control drawings, spacecraft/shroud static and dynamic envelopes, and specification lists.

3.2 Purpose

The primary purpose of interface control documentation is to assure establishment of compatibility among major assemblies supplied by different agencies and/or contractors. More specifically, it will:

(a) Define and illustrate details of physical and functional interfaces in terms of nominal values and allowable tolerances.

(b) Document and distribute on a timely basis coordinated design decisions to all affected program participators.

(c) Serve to preclude the occurrence of unilateral design changes in one system of an assembly that would affect other systems in the assembly across the interface.

3.3 Preparing Activity

3.3.1 Preparing Responsibility

Interface documentation will be prepared, maintained and distributed by the L/V System Contractor using information provided by all mission participators. It is the responsibility of both the S/C Center and Agena Project to ensure that all information relative to the interface and furnished to the L/V System Contractor is clear, concise, accurate, adequate, germane and timely.

3.3.2 First Release

The nominal sequence of events to transact an original set of approved interface documentation is as follows:

(a) The S/C Center transmits s/c interface information to the Agena Project and sends information copies to the L/V System Contractor concurrently.

(b) The L/V System Contractor prepares the interface documentation and distributes copies.

(c) Comments are compiled by the S/C Center and transmitted in writing or in the form of a marked up print to the Agena Project with information copies to the L/V System Contractor.

(d) When areas of disagreement are encountered or clarification of information is required, an interface meeting can be held to resolve matters.

(e) The L/V Systems Contractor prepares a revised version of the interface documentation and distributes same.

(f) The interface documentation is signed off by the S/C Center and Agena Project representatives at the next interface meeting.

3.3.3 Revisions

A. Revisions to interface documentation are to be made when equipment design changes necessitate a change to an interface or when it is mutually determined that clarification of stated requirements are necessary.

B. Revisions to the interface documentation can be proposed by either the Agena Project or the S/C Center. The nominal sequence of events to transact an approved revision is as follows:

1. The revision shall be proposed by one of the following means:

    a. Letter or TWX.

    b. Verbally, at an interface meeting, providing it is documented in the minutes of the meeting.

    c. Telcon, providing it is confirmed in writing within 3 working days.

2. Agreement is reached with respect to a revision between the S/C Center and Agena Project and it is documented. In most cases, the agreement and documentation of it will take place at an interface meeting.

3. Items (e) and (f) of para. 3.3.2 are then followed.

3.4 Distribution

The distribution of interface documentation shall be jointly established by the Agena Project and the S/C Center as soon as possible in the mission planning. Distributions shall be specified in the IPSD (see para. 4.0).

3.5 Schedule

Scheduling of Interface Control Documentation is handled via the IPSD (see para. 4.0).

4.0 Interface Plan and Schedule Document (IPSD)

4.1 Purpose

The IPSD will serve as the official plans and schedule document for all interface activities conducted during the s/c and L/V design, test and manufacturing phases of a mission. Established interface activities conducted at the launch base are not to be included in the IPSD. Goddard Launch Operations at WTR and ETR will coordinate the interface activities at the launch bases using established procedures (see para. 13.0). The IPSD shall not contain information regarding the launch vehicle and AGE engineering evaluation test programs, qualification test programs, launch base tests, etc. This information is provided in the Integrated Test Plan.

Examples of those interface activites and events to be included in the IPSD are:

(a) Matchmates

(b) Hardware exchanges

(c) Special interface tests

(d) Interface document generation, review and releases for such documentation as:

  (1) Requirements and Restraints Document

  (2) Control drawings

  (3) Program Requirements Document

  (4) Launch Operations Plan

(e) Studies

(f) Design Reviews

(g) Launch base efforts that require definition early in the mission planning, or are unique in nature. Examples are:

  (1) End-to-end calibration tests

  (2) Combined Systems Tests

  (3) Special matchmates

Note: The detailed scheduling of these launch base activities are, in general, the responsibility of GLO and only the submittal dates of the detailed schedules will be provided in the IPSD.

4.2 Format

The IPSD shall list the following information:

(a) Milestone Event

(b) Milestone Schedule

(1) Coordinated need date

(2) Expected date

(3) Completed date

(c) The organization(s) responsible for the event.

(d) NASA PERT number, if applicable.

(e) Special Considerations

In the case of interface tests, the IPSD will specify the conditions of the test; such as, purpose, test site, hardware requirements and special ground rules. A brief statement of the purpose and/or objectives of studies and documents will be included in this Special Considerations column together with the distribution list.

4.3 IPSD Preparation

4.3.1 Preparation Responsibility

The publication and distribution of the IPSD shall be performed by the Launch Vehicle System Contractor.

4.3.2 First Release

The nominal sequence of events to transact an approved IPSD is as follows:

(a) After the RRD first release, the S/C Center and the Agena Project shall submit to each other a list of milestones (with desired need dates) they wish incorporated in the document.

(b) Comments are to be compiled by the respective organizations and transmitted in writing to each other.

(c) At an interface meeting the milestone events and schedules to be included in the IPSD are finalized. The agreements reached shall be documented and signed by the S/C Center and Agena Project representatives.

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(d) The L/V System Contractor shall take the information generated under item (c) and publish the IPSD within ten working days.

4.3.3 Revisions

The nominal sequence of events to transact approved revisions is as follows:

(1) Same as (1) paragraph 3.3.3B.

(2) Same as (2) paragraph 3.3.3B.

(3) The L/V System Contractor shall issue revision sheets as approved changes are submitted by the Agena Project.

(4) The L/V System Contractor shall incorporate all revisions into a new issue of the IPSD when accumulation of revision sheets necessitates a new issue.

4.4 Coordinated Need Dates

No party is permitted to unilaterally change a coordinated need date.

4.5 Distribution

The IPSD distribution shall be jointly established by the Agena Project and the S/C Center and supplied to L/V System Contractor with the initial milestone information.

5.0 Studies

5.1 Study Definition

The term "study" as used herein includes (1) the basic analytical process performed in determining features of a hardware design or trajectory and (2) limited experimentation, including construction and implementation of bread-board type test hardware.

5.2 Study Implementation

Studies to define L/V characteristics as they apply to, influence, or constrain a mission are conducted by the Agena Project or L/V contractors responsible to the Agena Project. Other centers may request such studies of the Agena Project if they have a need in this regard. Study requests should not be submitted unless there is a clear need for the study and the following conditions prevail:

1. Study scope is such that the study effort can be clearly defined and any analysis required can be rigorously bounded and made finite.

2. Study is required to define mission requirements.

3. Study is required to define launch vehicle performance and configuration.

4. Study cannot be provided under the launch vehicle adaptation and integration contract due to schedule considerations.

5.3 Presentation of Study Results

(See para. 12.2 herein)

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6.0 Information Exchange

6.1 Requests for Transmission of Technical Data

1. Requests for technical information may be made by letter, TWX, telcon or at interface meetings. The channel for obtaining such information is to address such requests to the government agency sponsoring the particular contractor. Telcon requests shall be confirmed in writing within three (3) working days.

2. In emergencies, when the sponsoring agency cannot be contacted, requests for technical information may be made via telcons to the local government plant representatives. Such requests shall be confirmed in writing within three (3) working days; directed to the sponsoring government agencies, with information copies to the local government plant representatives.

6.2 Transmittal of Technical Data

1. Data transmissions may be made by letter, TWX, telcon or at interface meetings. The channel to be used for transmission of technical data is again to work through the sponsoring government agencies. Telcon data transmissions shall be confirmed in writing within three (3) working days.

   To expedite the receipt of written data, information copies may be addressed between contractors. It must, however, be recognized that any data received via this procedure is unofficial until approved by the sponsoring government agencies. Approval or disapproval for use of such information will be given on a timely basis.

2. In emergencies, the procedures similar to 6.1.2 above may also be employed in obtaining data.

6.3 Information Visits

Contractor to Contractor, Government to Government, Contractor to Government, and Government to Contractor information visits may take place between engineers representing any of the organizations involved in the project provided that:

(a) The information desired cannot be made available by other communication means.

(b) The visit is arranged and approved in advance via telcon or other communication by the sponsoring government agencies.

(c) The agenda of the meeting, the location, the date, the time, the personnel to be visited, and the attendees from each visiting organization be specified.

(d) Both sponsoring government agencies or their plant representatives are attendees.

(e) The number of visitors be kept to an absolute minimum.

Minutes of such meeting shall be prepared by the host organization and issued within ten (10) working days after the conclusion of the meeting. Copies of these minutes shall be distributed to all participating organizations.

7.0 Action Requests

Requests for action other than for information exchanges shall only be made by letter, TWX, and at official interface meetings that are documented. To allow proper control of the program, only the S/C Center and Agena Project personnel will accept action items. It will be the responsibility of the government agency accepting an action item to establish a schedule of completion.

8.0 Interface Meetings

8.1 Definition

A meeting between S/C Center and Agena Project engineering personnel to define and/or resolve the technical aspects of integrating a s/c and its mission objectives into the L/V system.

8.2 Purpose

Interface meetings will be convened to:

(a) Discuss engineering interface problems and formulate necessary action.

(b) Review and update the Interface Plan and Schedule Document.

(c) Permit direct and timely exchange of required technical information.

(d) Formulate recommendations to the Mission and Agena Project Managers.

(e) Monitor and review interface control drawings.

(f) Review and record results of technical data exchanges.

(g) Review the L/V Requirements and Restraints Document.

(h) Arrive at agreements with respect to interface problems (technical, planning, schedule).

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8.3 Meeting Chairman

Chairmanship of interface technical coordination meetings shall be the responsibility of the Agena Project Engineer or his alternate.

8.4 Meeting Attendees

The attendance by contractor and NASA personnel shall be kept to a minimum. The meeting will be limited to those personnel associated or directly concerned with specified agenda items. The following complement of attendees is, however, required before a meeting can convene.

1. The chairman or his alternate.

2. The S/C Center representative or his alternate.

3. The local government resident representative if the meeting is held at a contractor's plant.

Contractor engineering support at interface meetings shall be left to the discretion of each party.

8.5 Meeting Agenda

The meeting chairman shall solicit and coordinate agenda items from concerned agencies in collaboration with the S/C Center representative. The agenda, which is to be mutually agreed to by the S/C Center and Agena Project representatives, will be distributed to all participators at least five (5) working days prior to the meeting date.

8.6 Meeting Location

1. The meeting will be held at the location most appropriate for the agenda.

2. The selected host shall provide conference room(s) and administrative services as may be required.

8.7 Initiation

Interface meetings may be requested by either the Agena Project or S/C Center representatives but mutual agreement is required to hold the meeting. During the first meeting of the mission, arrangements may be made for the subsequent meeting and this procedure may be repeated for all following meetings.

8.8 Meeting Documentation

A. Action Items and Agreements

The S/C Center representative and the meeting chairman shall concur on each action items as to definition, assignment, and response date. Before the conclusion of each meeting, new action items, agreements and the resulting commitments shall be recapitulated by the chairman, documented and signed by the S/C Center and Agena Project representative. The approved action item list shall be distributed prior to the conclusion of the meeting. During each meeting, the action items established at previous meetings will be reviewed and closed out when the action has been completed. Those action items not closed out shall be listed as delinquent in the minutes of the meeting.

B. Meeting Minutes

The minutes of the meeting shall be prepared and distributed by the meeting host organization no later than ten (10) working days after the meeting. The minutes shall contain at least the following:

(1) A list of all attendees.

(2) Corrections to the minutes of the previous meeting.

(3) A status of open action items.

(4) A summary of any pertinent technical data exchanges that transpired during the meeting.

(5) If appropriate, the tentative date, place and agenda of the next meeting.

(6) A copy of the action items and agreements developed during the meeting.

(7) Delinquent action items that were assigned during previous meetings.

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8.9 Unresolved Problems

The chairman and the S/C Center representative will refer unresolved interface problems to the Agena Project Manager and/or the S/C Center Project Manager.

9.0 Interface Testing

9.1 Definition

Testing conducted for any purpose on a hardware assembly that contains systems, or components, supplied by more than one contractor/government agency so that the Test Contractor (contractor conducting the test) is directly or indirectly testing the hardware supplied by the Interfacing Contractor(s).

9.2 Exclusions

1. Matchmates are excluded from this definition and procedures for matchmates are provided in Section 10 herein.

2. Testing of hardware delivered by one contractor to another per government agreements for the exclusive use of the recipient contractor is excluded from this definition. In such cases, the supplying contractor does not concur in nor observe the testing of the hardware and, under these circumstances, cannot be held accountable for the performance of the hardware.

9.3 Schedule

The schedule of testing and other pertinent details are to be coordinated by means of the IPSD (Section 4.0 herein). The Spacecraft Center is advised to include the requirements for such testing in the RRD (Section 2.0 herein).

9.4 Test Plan

The Test Contractor is to prepare a Test Plan and issue it to the Sponsoring Government Agency 45 days prior to the start of testing. The Test Plan is to include such information as the purpose, location, equipment involved, number of tests and type, environment, duration, etc. Informational copies of the Test Plan are to be sent concurrently to the Associate Government Agency(s) and the Interfacing Contractor(s). Combined comments from the Associate Government Agency(s)/Interfacing Contractor(s) are to be sent to the Sponsoring Government Agency with an informational copy supplied to the Test Contractor(s) 30 days before the test. Concurrence or noncurrence of these comments by the Test Contractor are to be returned to the Sponsoring Government Agency 15 days before the test. In the event that comments by the Associate Government Agency(s)/Interfacing Contractor(s) are not approved by the

Sponsoring Government Agency/Test Contractor(s), it will be the responsibility of the Sponsoring Government Agency to resolve the issues to the satisfaction of all parties involved.

Note: Tests are not to be delayed for receipt of comments on the Test Plan from the Associate Government Agency(s)/Interfacing Contractor(s).

9.5 Witnesses

The Associate Government Agency(s) and Interfacing Contractor(s) are expected to attend. These parties are to provide a list of expected attendees to the Sponsoring Government Agency and Test Contractor within 5 days prior to the test.

Note: The Test Contractor will exert every effort to keep witnesses informed of the estimated times at which tests will commence. However, the interface test will not be delayed for the arrival of any witness.

9.6 Conduct of Witnesses

Witnesses are not to direct or interfere with the scheduled test unless the test is detrimental to their supplied hardware. If such is the case, or for any other reason, witnesses may appeal to the Sponsoring Government Agency to delay said test until the problem in question is resolved.

9.7 Test Reports

The Test Contractor is to release a preliminary report as soon as practicable and follow up with a final report. These reports are to be sent to the Sponsoring Government Agency with informational copies to the Associate Government Agency(s) and Interfacing Contractor(s).

10.0 Matchmate

10.1 Definition

Matchmate is a compatibility test conducted with flight hardware to demonstrate proper mechanical and electrical mating of flight hardware.

10.2 Schedule

The matchmate schedule and other pertinent details are to be coordinated by means of the IPSD (Section 4.0 herein). The Spacecraft Center is advised to include the requirements for such matchmating in the RRD (Section 2.0 herein).

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10.3 Master Procedure

An interface meeting is to be called by the LeRC Agena Project about 60 days prior to the matchmate date with all participating agencies or contractors in attendance. All parties actively participating in the physical matchmate or the supplying of hardware are responsible for presenting written information and/or drawings on their activity or hardware at the meeting. Based on the discussion of the meeting and information supplied, the Vehicle Integration Contractor will prepare and issue a Master Procedure 40 days prior to the matchmate date. The Master Procedure will contain a listing of the hardware, the location of the matchmate, the responsibility of each participating agency or contractor, the sequence of operations to be followed in performing the matchmate, and other pertinent information.

10.4 Approval of Master Procedure

All participating parties are to approve the Master Procedure in writing 20 days prior to the matchmate date and send their approvals to LeRC and the Vehicle Integration Contractor with informational copies to all other participating parties.

If any participating Government Agency desires changes in the Master Procedure, LeRC will resolve the differences by telcon and follow up with a letter confirmation to all participators or arrange for another interface meeting, if required.

The Vehicle Integration Contractor is to publish a final version of the Master Procedure, if changes are required, 10 days prior to the matchmate date.

Note: Matchmate shall not occur until the Master Procedure is approved.

10.5 Participation

All Associate Contractors supplying hardware to the matchmate will be active participators. Quality Assurance representatives from all participating contractors are to be in attendance. A list of expected attendees from all organizations is to be sent to the Host Contractor 5 days prior to the matchmate date.

10.6 Responsibilities

Each participating contractor is solely responsible for his hardware during shipment and is responsible for either performing or directing the handling of his hardware during all phases of the matchmate.

The Host Contractor is responsible for supplying the proper environment and standard handling equipment for the matchmate.

10.7 Location

The location and the Host Contractor for the matchmate is to be mutually agreed on by the parties involved based on the hardware to be tested and other circumstances of the matchmate.

10.8 Test Report

The Vehicle Integration Contractor is to release an evaluation/ corrective action letter report. This report is to be sent to LeRC with informational copies to participating Government Agencies and Associated Contractors.

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11.0 Launch Vehicle Design Reviews

11.1 Purpose

The purpose of the L/V Design Review is to present to the S/C Center, the Agena Project, and NASA Headquarters, the L/V design, testing, manufacturing and planning status for a particular mission so the NASA personnel can make their own detailed evaluation of the L/V effort.

11.2 Presentation

The Design Review presentation is made by the L/V System Contractor and/or the L/V Associate Contractors, as appropriate. Standard practice is for three reviews to be held for each mission at the LMSC plant in Sunnyvale, California. The first review is held after initial design concepts have been established, the second or "detailed" is held at some appropriate interim period, and the third or "final" is held just prior to buy-off of the vehicle and attendant mission peculiar equipment by the government.

11.3 Attendees

The attendees of the Design Review are interested NASA, Air Force, and Contractor representatives associated with the particular project. Each government organization shall forward a list of attendees, which includes their Associate Contractors' representatives, to LMSC/SV five (5) days prior to the review.

11.4 Spacecraft Center Responsibilities

It is the responsibility of the S/C Center personnel to carefully and critically review the design information presented to ensure that all designs are compatible with and have the performance characteristics to meet the s/c requirements and objectives. Areas of incomplete information or technical doubt should be noted and additional discussion requested. All parties to the review have a responsibility to make known any objections or concerns they may experience as a result of the review. These concerns should be stated at the review or provided in writing to the Agena Project within a period of one week subsequent to the review.

11.5 Agenda

Government agencies desiring inclusions of particular agenda items in the review shall submit their request to the Agena Project at least four weeks prior to the Design Review. These agenda items will be coordinated by the Agena Project and transmitted to the L/V System Contractor and/or Associate Contractors. These contractors will include in their presentations a discussion of all agenda items so received from the Agena Project. It should be clearly understood, however, that the Design Review pertains to the L/V System. If requested agenda items are not germane to the review, they will not be included on the agenda but will be handled at some subsequent meeting or via other appropriate action.

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12.0 Other Meetings

12.1 Post Flight Review

12.1.1 Initiation

The Post Flight Review may be waived if conditions warrant. After a flight has been completed, the Agena Project determines whether or not a Post Flight Review is needed.

12.1.2 Purpose

The purpose of the L/V Post Flight Review is to present to the Agena Project and other concerned organizations a detailed description of the launch and flight with emphasis placed on flight data evaluation and an explanation of any anomalies that occurred. The written Flight Evaluation and Performance Analysis Report is published by the L/V System Contractor about 45 days after launch. The Post Flight Review meeting, held 20 to 30 days after launch, provides an opportunity to assess the performance of the L/V System prior to the time the written report is issued so as to indicate specific areas to be thoroughly covered in the report and, in the case of follow-on launches, to determine the actions necessary to improve mission executions or correct anomalies.

12.1.3 Presentation

The Post Flight Review presentation is made by the L/V System Contractor and Associate Contractors at the LMSC plant in Sunnyvale, California. Under unusual circumstances, the review may be held at other locations.

12.1.4 Attendees

Each organization shall determine their attendees and forward a list of same to LMSC/SV five days before the review.

12.1.5 Agenda

Government agencies desiring inclusions of particular agenda items in the review shall submit their request to the Agena Project at least two weeks prior to the Post Flight Review. These agenda items will be coordinated by the Agena Project and transmitted to the L/V System Contractor and/or Associate Contractors. These contractors will include in their presentations a discussion of all agenda items so received from the Agena Project. It should be clearly understood, however, that the Post Flight Review pertains to the L/V system. If requested agenda items are not germane to the review, they will not be included on the agenda but will be handled at some subsequent meeting or via other appropriate action.

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12.2 Study Review

12.2.1 Presentation of Results

The manner of presentation of results from studies performed under paragraph 5 is at the discretion of the Agena Project. The results from many studies are presented only in the form of written reports. When warranted, the Agena Project follows the following sequence of presentation:

1. Preliminary presentation of results (usually midway through the study or upon completion of a significant segment of the study).

2. Final oral presentation of study results.

3. Written report submitted within 30 days after the final oral presentation.

12.2.2 Attendance at Study Reviews

Study review are normally held at the LeRC. The government organization initiating the study request of paragraph 5.0 will be invited to attend the final oral presentation, if held, and will receive copies of the written report. S/C Center contractor participation in the oral presentations and receipt of the written report will be at the discretion of the participating government agencies.

12.3 General Considerations

1. Contractors must have the consent of the responsible government agency(ies) to participate in a meeting.

2. Host organizations provide secretarial help and publish meeting minutes and action item lists.

3. Action item lists have no official status unless signed by a designated government representative from the government organization upon which an action has been placed.

4. Meeting minutes and action item lists should be prepared and distributed within 10 days subsequent to a meeting.

5. Meetings will not be called unless a formal agenda has been coordinated with all participating groups and distributed at least one week prior to the meeting.

13.0 Launch Base Activities

13.1 Launch Base Working Groups

The Launch Operations Working Group at ETR and the Launch Test Working Group at WTR are the official organizations for planning, coordinating and scheduling of day-to-day activities at the launch base. The NASA/GLO, participating launch service contractors, AFETR and the Mission Center comprise the membership of the LOWG. The 6595th Aerospace Test Wing, NASA/GLO, participating launch service contractors, and the Mission Center comprise the membership of the LTWG. The LOWG and LTWG are chaired by NASA/GLO and the 6595th ATW, respectively.

13.2 Launch Base Modifications - AGE

13.2.1 General

The mission peculiar launch base AGE modifications have two distinct phases; Phase I is Design and Manufacturing and Phase II is Installation and Checkout. The mission peculiar modifications are designed and manufactured by the Agena Contractor (LMSC in Sunnyvale) under contract to LeRC and the Booster Contractor (GD/C at San Diego for the Atlas and Douglas at Santa Monica for the Thor/TAT) under contract to the AF/SSD. The spacecraft AGE requirements to be met by vehicle contractors are to be provided in the L/V System Requirements and Restraints Document (section 2.0 herein). The actual installation and checkout (Phase II) of the modifications at the launch base is performed by different divisions of the same companies under separate launch services contracts. The installation of s/c AGE built by the S/C Center is also performed under these launch services contracts.

13.2.2 Integration

13.2.2.1 Planning

When the extent of mission peculiar modifications are small in scope, Agena Project personnel perform the necessary integration between the vehicle contractors and the S/C Center for the Phase I activity.

In the case where the peculiar modifications for a mission are of major scope, the Agena Project contracts with the L/V Contractor for the necessary integration between the vehicle contractors and the Spacecraft Center. Toward this end, the L/V Contractor produces an Activation Manual which provides some, or all, of the following:

(1) A definition of the scope of effort that includes responsibilities of all organizations concerned and a schedule for implementing the Activation Manual.

(2) An integrated milestone schedule which provides a graphical presentation of all major events and time spans of major tasks required.

(3) A conversion plan which provides a written description of all procedures, equipment, and tasks required for the activation of the launch base.

(4) An integrated activation schedule. This schedule is a detailed day-to-day working schedule for all Associate Contractors and the S/C Center and covers the detailed tasks required in the installation and checkout of ground support equipment.

The Activation Manual is jointly concurred in by the Agena Project, all Associate Contractors and the S/C Center.

13.2.2.2 Implementation

The implementation of launch base modifications is overseen by NASA/GLO for the Agena Project using the Activation Manual as a guide. The launch base working groups are the official means used by GLO for carrying out this integration.

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13.3 Launch Base Modifications - Facilities

All new L/V facilities or L/V facility modifications required at the launch complex in support of a mission are within the purview of the Agena Project. The S/C Center requirements associated with L/V facilities are to be specified in the L/V Requirements and Restraints Document (section 2.0 herein).

Mission facility design criteria are generated by all organizations and integrated into working documentation by a vehicle contractor. Implementation of facility changes based on these criteria may be through KSC, PLOO, C of E, or individual contractors depending upon the nature and scope of the work.